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SOFT LANDER PART VI

Mars Soft Lander Capsule Study (Entry From Orbit) - Conceptual Design III AND IV

Prepared by

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4.3 Concept III

The 1603 lb Concept III capsule is designed to perform the minimum science mission for an extended period. This mission consists of entry and surface atmospheric measurements, post-landed imaging and soil composition measurements with relay transmission of ten million bits of data per day early in the mission and whenever the orbiter is available for the relay. A low rate S-band system communicates data at other times during the mission lifetime of at least 90 days. Nominal environmental constraints and mission flexibility have been assumed as the basis for this conceptual design. The aeroshell diameter is 11.5 ft.

The capsule mission includes a boost and Earth orbit phase, an interplanetary phase, and a Mars orbit phase. The capsule then separates from the orbiter, performs a deorbit maneuver, descends and enters the atmosphere. Trajectory parameters for a typical mission are presented in Table 4.3-1. A parachute and a three-engine throttleable monopropellant propulsion system provide the final deceleration. After landing, the thermal environment is maintained by a combination of isotope and electrical heaters, a phase-change heat sink, and insulation.

Prior to deorbit of the capsule, all critical systems are sequentially activated for test under the control of the central computer and sequencer (CC&S). A first level systems test will be performed which is comparable to those which can be performed after the insertion of the capsule into the sterilization canister. These tests begin several hours before the actual separation sequence and include checkout of such systems as the landing and altimeter radars, the guidance system, the telecommunications, and the propulsion system. Data obtained during this test is transmitted from the capsule by means of the capsule relay transmission system to the orbiter data system for a subsequent transmission to the Earth. These tests and separation of the forward portion of the sterilization canister are performed in sufficient time to allow analysis of the results by ground operations personnel and transmission of corrective commands or memory updating information to the capsule before deorbit. These tests culminate in the full activation of all capsule systems

TABLE 4.3-1

NOMINAL TRAJECTORY

• LAUNCH	
DATE	19 SEPT 1973 (TYPE I)
VIS VIVA ENERGY, C_3	44.06 KM^2/SEC^2
• ARRIVAL	
DATE	17 MAY 1974
FLIGHT TIME, T_F	240 DAYS
HYPERBOLIC EXCESS VELOCITY, V_{HP}	2.86 KM/SEC
• ORBIT (POSIGRADE NORTH)	
SIZE	SYNCHRONOUS (1000–33 124 KM)
INCLINATION, i	60°
ASCENDING NODE, Ω	291.3° (EQUATORIAL, MARS EQUINOX)
ARGUMENT OF PERIAPSE, ω	114.5
INSERTION VELOCITY INCREMENT, ΔV	2.1 KM/SEC
PERIAPSE ROTATION ANGLE, ρ	66°
• DEORBIT	
TRUE ANOMALY, θ_{DO}	214°
VELOCITY INCREMENT, ΔV	361.3 FT/SEC
DEFLECTION ANGLE, δ	+ 25.7°
DESCENT TIME	3.4 HRS
• ENTRY (800 000 FT ALTITUDE)	
VELOCITY, V_e	15 110 FT/SEC
FLIGHT PATH ANGLE, γ_e	–16°
TRUE ANOMALY, θ_e	331.6
CAPSULE LEAD ANGLE, $\Delta\theta_e$	5.35°
• TERMINAL DECELERATION	
PARACHUTE DEPLOYMENT ALTITUDE	23 000 FT
ALTITUDE AEROSHELL IMPACT (VM-7)	5000 TO 6000 FT
TERMINAL PROPULSION INITIATION	6500 FT
• LANDING	
SOLAR ANGLE, Γ_S	60°
LATITUDE, δ	10°N
CENTRAL ANGLE FROM ENTRY	16.8°
TIME FROM ENTRY	440 SEC
LANDING TRUE ANOMALY	348°
ORBITER TRUE ANOMALY	348°
POST-LANDED VIEW TIME (34° GROUND SLOPE)	300 SEC

immediately involved in the deorbit and descent operations and include the alignment of the capsule inertial reference unit.

The capsule is released pyrotechnically from the orbiter. The capsule attitude control jets provide sufficient impulse to achieve the desired physical separation between the capsule and orbiter prior to deorbit ignition. During this period, the capsule inertial reference unit is commanded by the computer to rotate to the correct attitude for the deorbit maneuver. During this phase, attitude control limits of 0.25 degrees are maintained. Transmission of engineering data from the capsule during this and subsequent phases is achieved by means of relay transmission to the orbiter at a rate of approximately 4000 bits per second.

After achieving the correct separation from the orbiter and the proper attitude for the deorbit maneuver, ignition is commanded by the sequencer. The velocity increment is measured by the longitudinal accelerometer in the guidance and control system, and the deorbit motor shutdown is initiated by the guidance computer when the required velocity is attained. A backup command for motor shutdown is provided by the sequencer at the end of a preset time interval. The magnitude and direction of the deorbit maneuver are chosen to provide the desired entry flight path angle and to insure that at the time of entry the orbiter lags the capsule by 5° true anomaly.

Immediately after the deorbit maneuver, the capsule guidance system initiates a maneuver to place the capsule in the proper inertial attitude for the entry phase. After this is accomplished, attitude control is operated in a 3° deadband limit mode. The gyros continue to operate in an attitude hold mode. The capsule holds this attitude throughout the subsequent 3 to 10 hour descent period. Relay transmission of engineering parameters associated with the capsule environment and guidance equipment is continued.

At an altitude of approximately 800 000 feet (preprogrammed time initiation) the sequencer activates critical capsule entry systems. This includes activation of the radar altimeter, entry science instruments, and a change in the capsule telemetry data format for entry. At the point of sensible atmospheric contact, .05g longitudinal deceleration, the control transfers to a rate damping mode in the pitch and yaw axes to insure stability during entry. The roll axis control continues to operate in an attitude hold mode to insure proper orientation of the radar altimeter antenna.

Shortly after entry, ionization blackout may interfere with relay communications for a period as long as 150 seconds. To insure the full recovery of data obtained during this period, a core data storage buffer is used to delay all engineering and science data by an amount equal to this expected blackout duration. In this manner, data is transmitted both in real time and delayed time to insure full recovery of all data. The radar altimeter will normally begin data acquisition at an altitude of approximately 200 000 ft. The marking signal from the radar altimeter will initiate parachute deployment at an altitude of approximately 23 000 feet. The attitude control system is deactivated at this point. The aeroshell is released from the lander shortly after parachute deployment and allowed to fall clear of the vehicle. At this point the landing radar system is enabled; radar acquisition may occur, but its continuous operation is not required prior to parachute release. The parachute descent phase is chosen to be sufficiently long to insure that near terminal velocity is achieved by the parachute, and to insure surface impact of the aeroshell prior to the time that landing radar operation is required.

During this period the guidance and control system is operated in a soft caging mode. This provides a best estimate of the true flight path direction and averages the effect of natural or gust induced parachute oscillations. The terminal descent engines are ignited to stabilize the vehicle and the parachute is then released on the basis of the marking signal from the radar altimeter at an altitude of approximately 6500 feet. The guidance and control system then commands the vehicle to the previously determined nominal flight path direction to provide a favorable orientation for landing radar acquisition.

After landing radar acquisition occurs, the vehicle is oriented to align the longitudinal axis with the velocity vector as determined by the landing radar. During this phase primary attitude control information is derived from the radar and the inertial reference unit provides attitude rate damping to the system. The vehicle holds this attitude and the engines are throttled to provide the constant .8g acceleration as measured by the longitudinal accelerometer in the guidance system.

The vehicle continues in this mode until intercept of a velocity-altitude control line which is stored in the guidance computer. At this point the engines are throttled to provide the required acceleration to keep the vehicle on this flight profile. Control information in this phase is based on landing radar measured slant range and total velocity. It is used to compute attitude and acceleration commands to the engines. Actual vehicle acceleration is measured by the longitudinal accelerometer. Throughout this phase, the vehicle maintains an attitude which places the total velocity vector coincident with the roll axis. The vehicle continues down this flight path to a point of approximately 50 ft altitude at a velocity of 10 ft per sec. At this point, the inertial attitude hold control is initiated and a constant 10 ft per sec descent phase, as measured by the radar, is begun. At an altitude of approximately 10 ft, engine thrusting is terminated and the vehicle descends to the surface.

Immediately after landing, the surface payload is activated to begin a transmission of surface science data to the orbiter. The entry trajectory is selected so that the orbiter is approximately overhead at the time of landing. This condition provides approximately 4 minutes of useful communication time from the lander to the orbiter before loss of visibility occurs. This time is sufficient to permit transmission of 6.4×10^6 bits of data including four wide angle images from the low resolution facsimile camera, together with initial measurements of surface meteorological conditions. This data transmission occurs at a rate of 41 600 bits per second.

After completion of this transmission, the sequencer terminates transmitter operation and the payload systems begin normal data acquisition operations. This includes deployment of the alpha spectrometer for a calibration measurement cycle. The meterological measurements of temperature, pressure, humidity and wind are made every 15 minutes. This data is then stored in a core memory for later transmission. After the alpha spectrometer has completed its calibration measurement, this data is transferred to core storage. The alpha spectrometer is then lowered to the surface to allow it to make an 8 hour surface composition measurement. The cycle of meteorological measurements continues for one full day with data being accumulated in the core storage.

The next data transmission period is initiated and terminated by sequencer command and occurs at the overflight of the orbiter near its periapse point, approximately 24.6 hours after landing. However, the uncertainty in the time of orbiter visibility is greater than the 4 minutes required for data readout. Therefore, all stored data, as well as facsimile data, is transmitted completely every 4 minutes for approximately 20 minutes. This insures acquisition of one complete set of data (10^7 bits) by the orbiter during its overflight. During this transmission interval, the facsimile camera is operated continuously and its data is interleaved with data from the core memory. Data read from the core memory is restored in the memory to allow repetitive transmission of the same data. Data transmitted to the orbiter is stored on a tape recorder mounted in the orbiter for later direct transmission to Earth.

This cycle is repeated until the orbiter mission requirements cause it to be diverted and no longer available for relay. If the orbiter is in view on subsequent orbits, the relay transmission mode may again be activated. During the remaining portion of the mission, a low-rate S-band transmits engineering and meteorological science data directly to Earth. An Earth-to-capsule command link operates for two hours each day for sequence modification. Receiver operation is limited to two hours per day to conserve electrical energy.

The capsule is designed for a lifetime of 90 days while operating in the nominal mode, but actual lifetime may exceed this and be terminated by one of the following phenomena:

- o Failure of batteries designed for charge/discharge cycling for only 90 days.
- o Degraded output of solar cells because of dust settling or (for southerly landing sites) increasingly poor solar incidence angles.
- o Equipment failure.

The probability of mission success depends on the reliability of the equipment to perform in the design environments and the capsule's invulnerability to off-nominal conditions which may be encountered. The Concept III design does not include redundant components. This design approach was selected to facilitate concept comparison and to ultimately assure the most efficient approach to providing a high reliability capsule. In using a similar approach for the VOYAGER Capsule Phase B design (see NASA CR 89672), it was determined that a gross weight increase of 73 lbs for redundant components increased reliability for performing all functions to .71. A .71 reliability for the Concept III capsule could be achieved by adding redundant components which would increase the capsule weight about 56 lbs. This increased capsule weight would require an increase in capsule diameter to 11.8 ft to assure a deployment Mach number of 2.0.

The Concept III capsule is designed to survive the nominal environment. It is vulnerable, however, to clouds and dust which could reduce the efficiency of the solar cells. This would result in an abbreviated surface lifetime; however, power for a minimum mission duration of one day is available from the batteries alone.

4.3.1 SYSTEM CONFIGURATION - The flight capsule consists of three major components:

- Capsule
- Sterilization Canister
- Adapter

The capsule, after separating from the orbiting flight spacecraft delivers the entry science into the Martian atmosphere and the lander to the surface of Mars. The interior arrangement of the capsule is shown in Figure 4.3.1-1; it includes equipment for:

- Deorbit
- Attitude control
- Entry
- Descent
- Landing

The Sterilization Canister encloses the capsule and protects it against recontamination from terminal sterilization until separation in Martian orbit.

The adapter supports the Capsule and Sterilization Canister for all ground handling, launch, and flight loads.

A general arrangement of the flight capsule is shown in Figure 4.3.1-2 and its relationship to the launch vehicle shroud (12.98 ft. diameter) is shown in Figure 4.3.1-3.

The four basic structural modules that make up the flight capsule are the sterilization canister, adapter, aeroshell, and lander (See Figure 4.3.1-4). These four structural modules, plus installed support systems, make up the total flight capsule.

4.3.1.1 <u>Sterilization canister</u>. - The canister consists of two structures, each semi-hemispherical in shape. Each structure is of aluminum semi-monocoque construction, stiffened by rings and external stringers. The Forward and Aft canisters are joined at the major diameter plane by a contained explosive separation device. This device, which is redundant, is sandwiched between the bolt flanges on the Forward and Aft canisters. A half inch blanket of multilayer

FLIGHT CAPSULE INTERIOR ARRANGEMENT

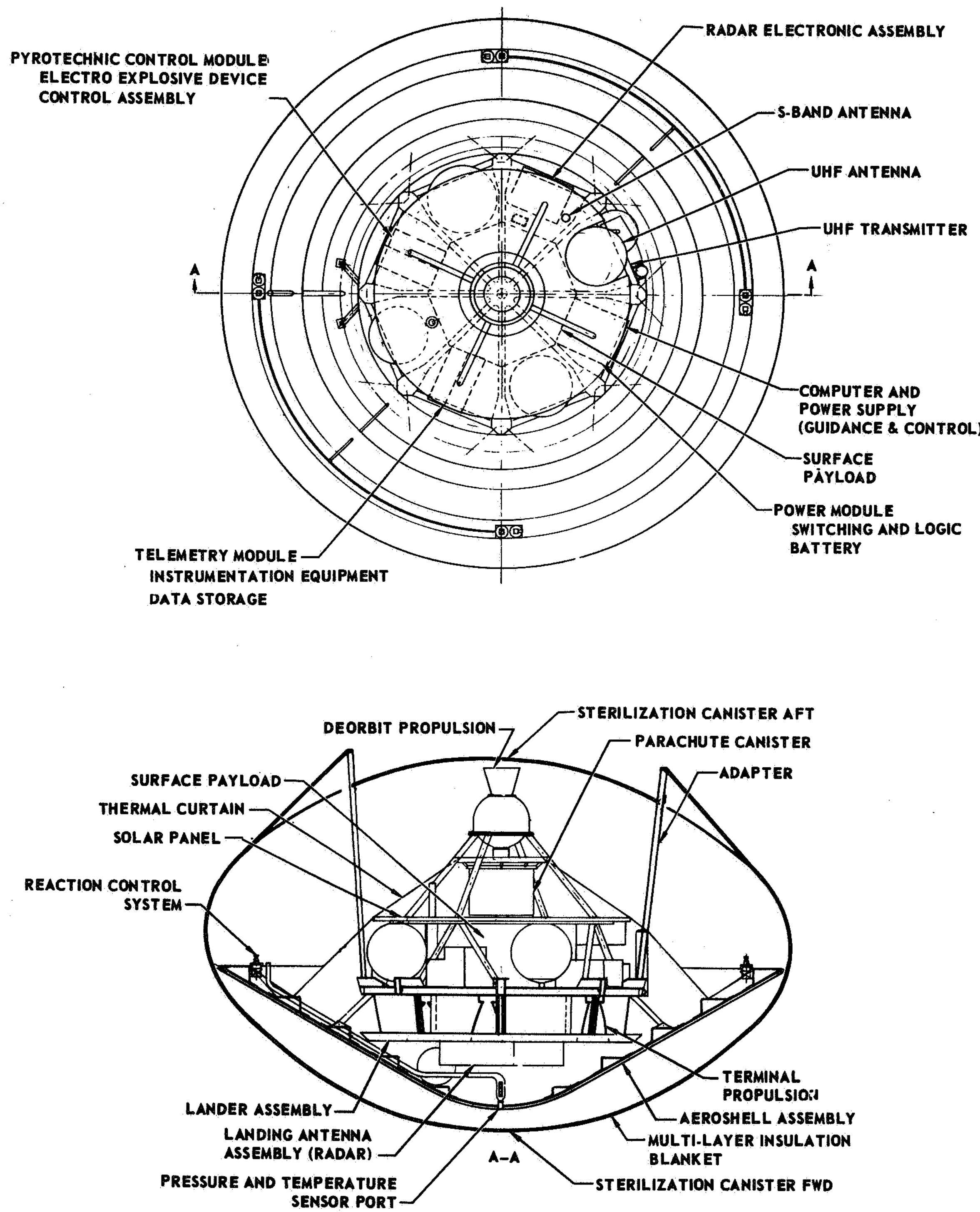


FIGURE 4.3.1-1

FLIGHT CAPSULE
GENERAL ARRANGEMENT
CONCEPT III

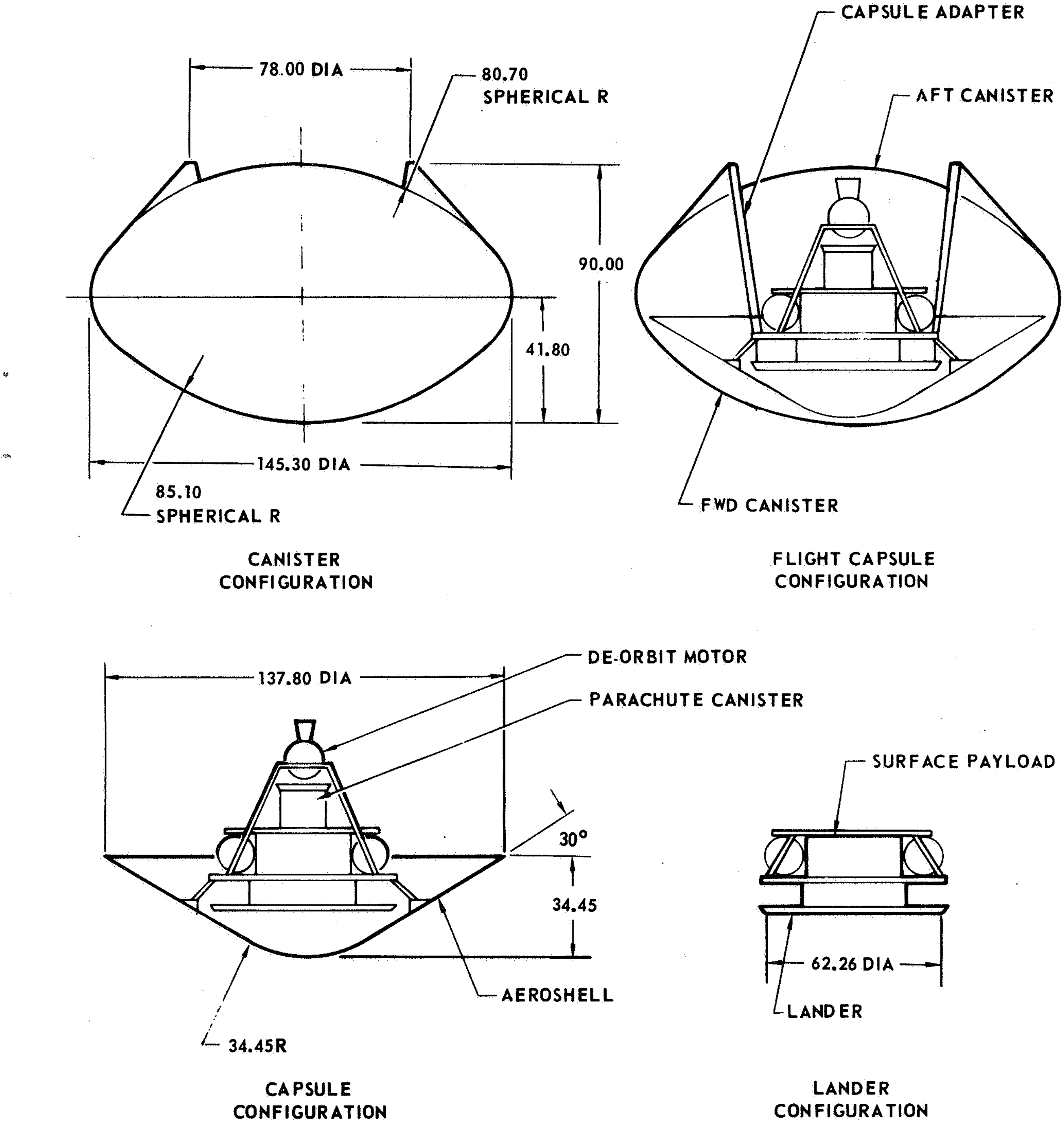


FIGURE 4.3.1-2

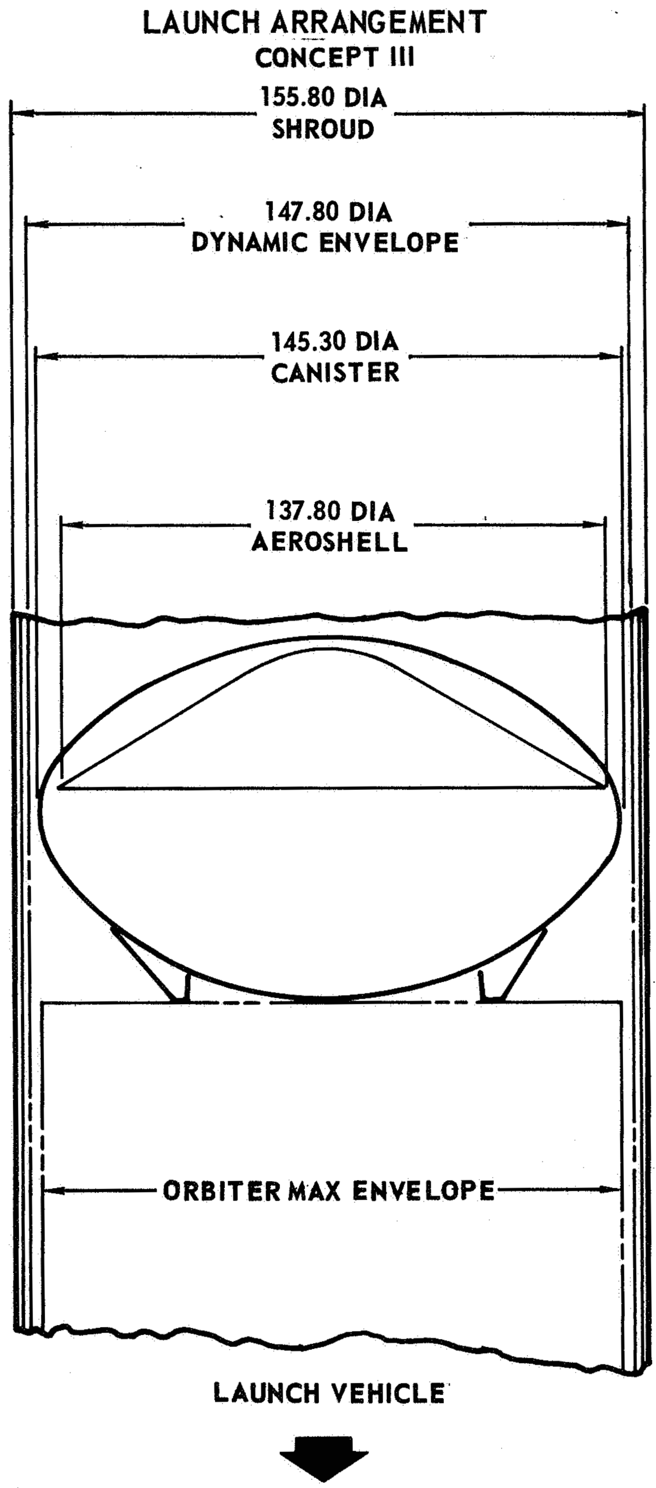

LAUNCH ARRANGEMENT
CONCEPT III
155.80 DIA
SHROUD
147.80 DIA
DYNAMIC ENVELOPE
145.30 DIA
CANISTER
137.80 DIA
AEROSHELL
ORBITER MAX ENVELOPE
LAUNCH VEHICLE

FIGURE 4.3.1-3

FLIGHT CAPSULE BASIC MODULES

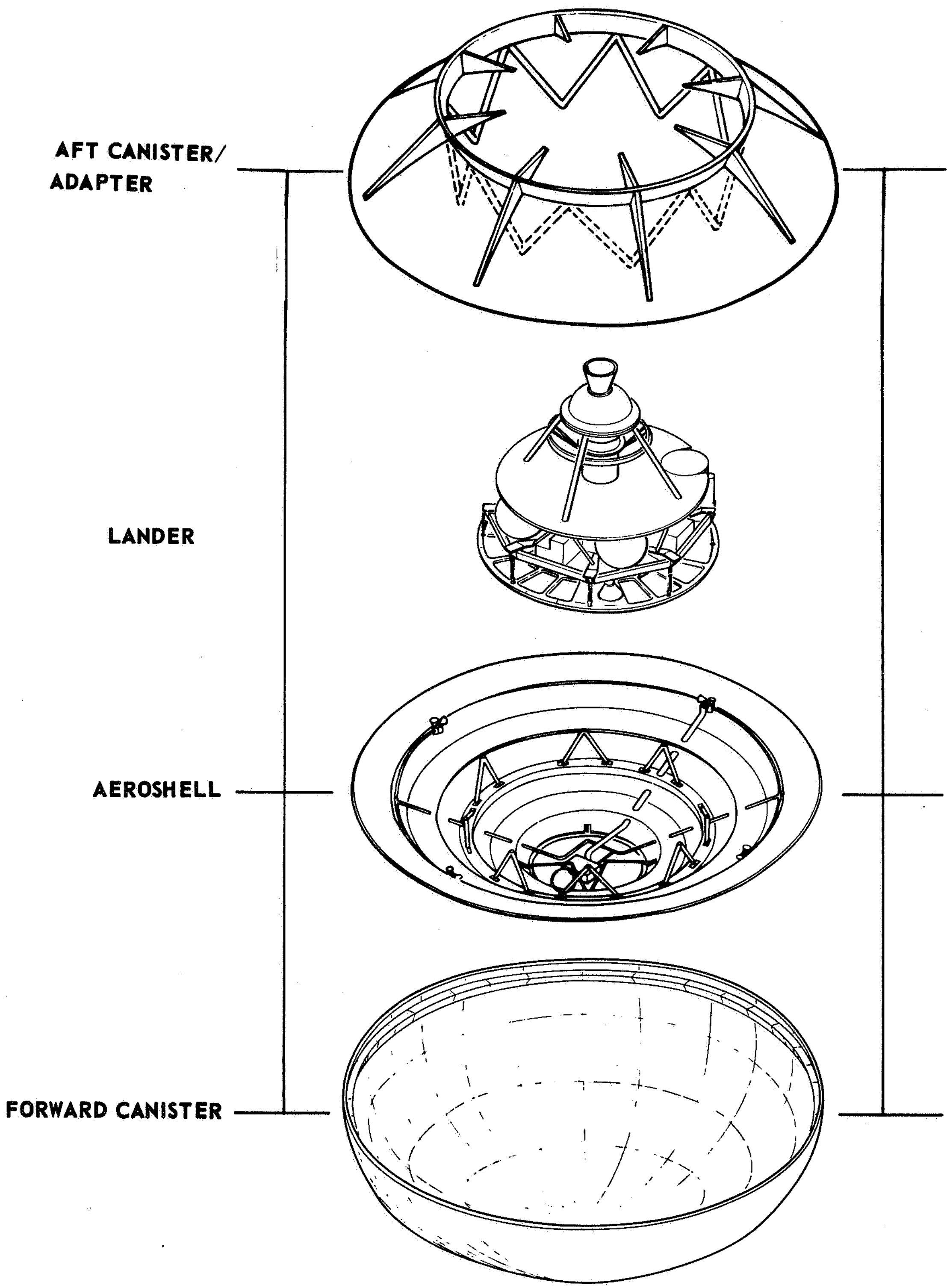


FIGURE 4.3.1-4

insulation is installed on the external surface of both canister sections, with a joint at the canister separation plane.

Equipment installed in the canister, in addition to the release separation device includes:

- Pressurizing and Venting
- Electrical Power
- Inflight Checkout Programmer
- Command Decoder
- Telemetry/Instrumentation

The pressurizing and venting subsystem consists of redundant vent valves that vent the canister during ascent and are closed by redundant differential pressure switches; redundant evacuation valves to vent the canister during sterilization and during space flight; and two relief valves. Biological filters are installed in each valve to prevent recontamination through an open valve.

The structure of the canister assembly protects the capsule from recontamination after leaving the sterilization chamber. It completely encloses the capsule to form a sealed wall against microbial contamination.

4.3.1.2 Adapter. - The adapter is made up of 16 tubular aluminum truss members, welded in a continuous ring. Members alternately converge in a zigzag pattern at eight attach points on the capsule bus and at eight attach points on the aft canister. The adapter is attached to the capsule by eight explosive bolts.

The adapter supports the capsule for all ground handling, launch, and flight loads. The adapter is designed to install the capsule within the canister without access doors in the aeroshell or in the canister.

The only equipment mounted on the adapter is an image antenna and stub antenna. These are used to check out the capsule subsystems.

4.3.1.3 Aeroshell. - The aeroshell is a single-faced, longitudinally corrugated titanium structure with closed triangular rings. The nose structure contains an instrument head which collects gas samples and measures pressure and temperature. The remainder of the nose cap is covered externally with a non-ablative material (See Figure 4.3.1-5).

The conic segment of the aeroshell and the upper ring are covered externally with an ablating material. This material is a foamed reinforced methyl phenyl silicone, filled into honeycomb, and hard bonded to the structure.

The supports for mounting the lander in the aeroshell are installed on a structural ring. Eight bipods are attached to this ring at their base and to the lander structure at their apex. Attachment at the ring includes torison springs to rotate the bipods outward upon lander release. Attachment at the lander is by explosive bolts at four points and compression preloaded sockets at four other points.

The reaction control system is installed in the aeroshell. It is a cold gas pressurized system consisting of eight thrusters, four for roll control, and four for pitch and yaw control. The thrusters are mounted on the base ring at four places, 90° apart. Regulators, valves, filters, tank, etc. are located in the nose area below the lander. The subsystem operation begins with the separation of the capsule from the orbiter. Firing commands to the subsystem provide attitude control during orientation, coast, deorbit motor firing, and atmospheric entry. During entry into the Mars atmosphere, attitude rate damping is provided.

4.3.1.4 Lander. - The lander is composed of three structural elements: the footpad, the shock attenuating ring, and the base platform. (See Figure 4.3.1-6).

The footpad is made of titanium skin, radial beams, and rings. Cutouts in the lower surface allow for firing the terminal descent engines. Three stabilizing legs are mounted flush in the footpad and are spring actuated and mechanically locked to stabilize the system after landing. The landing radar antenna assembly is attached to the lower surface and is crushed on impact.

The impact attenuator is a cylinder of crushable aluminum honeycomb, such as "Trussgrid"; it is sandwiched between the footpad and base platform and crushes on impact. Eight pulley assemblies, equally spaced around the periphery,

AEROSHELL GENERAL ARRANGEMENT
CONCEPT III

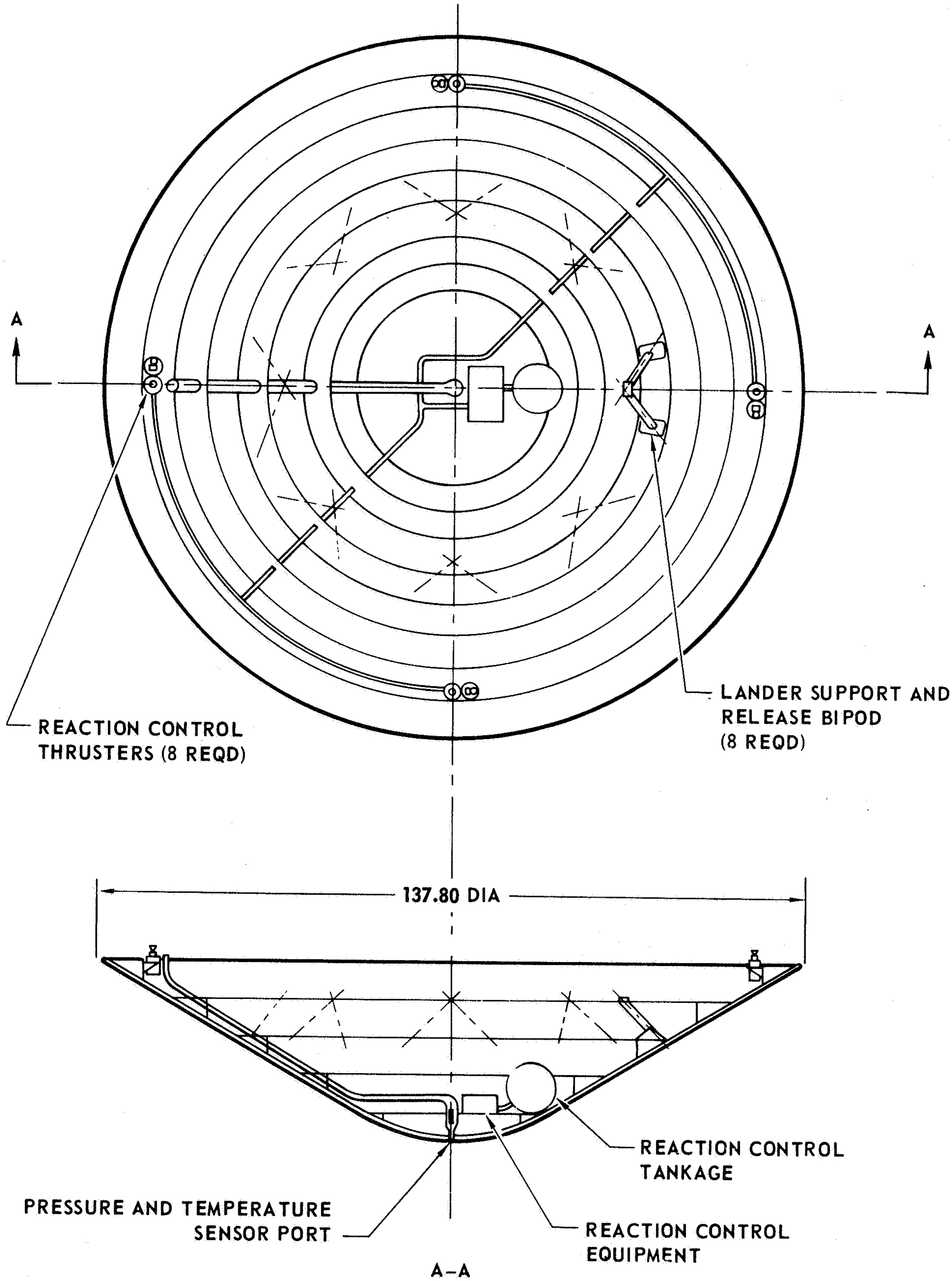


FIGURE 4.3.1–5

LANDING SYSTEM

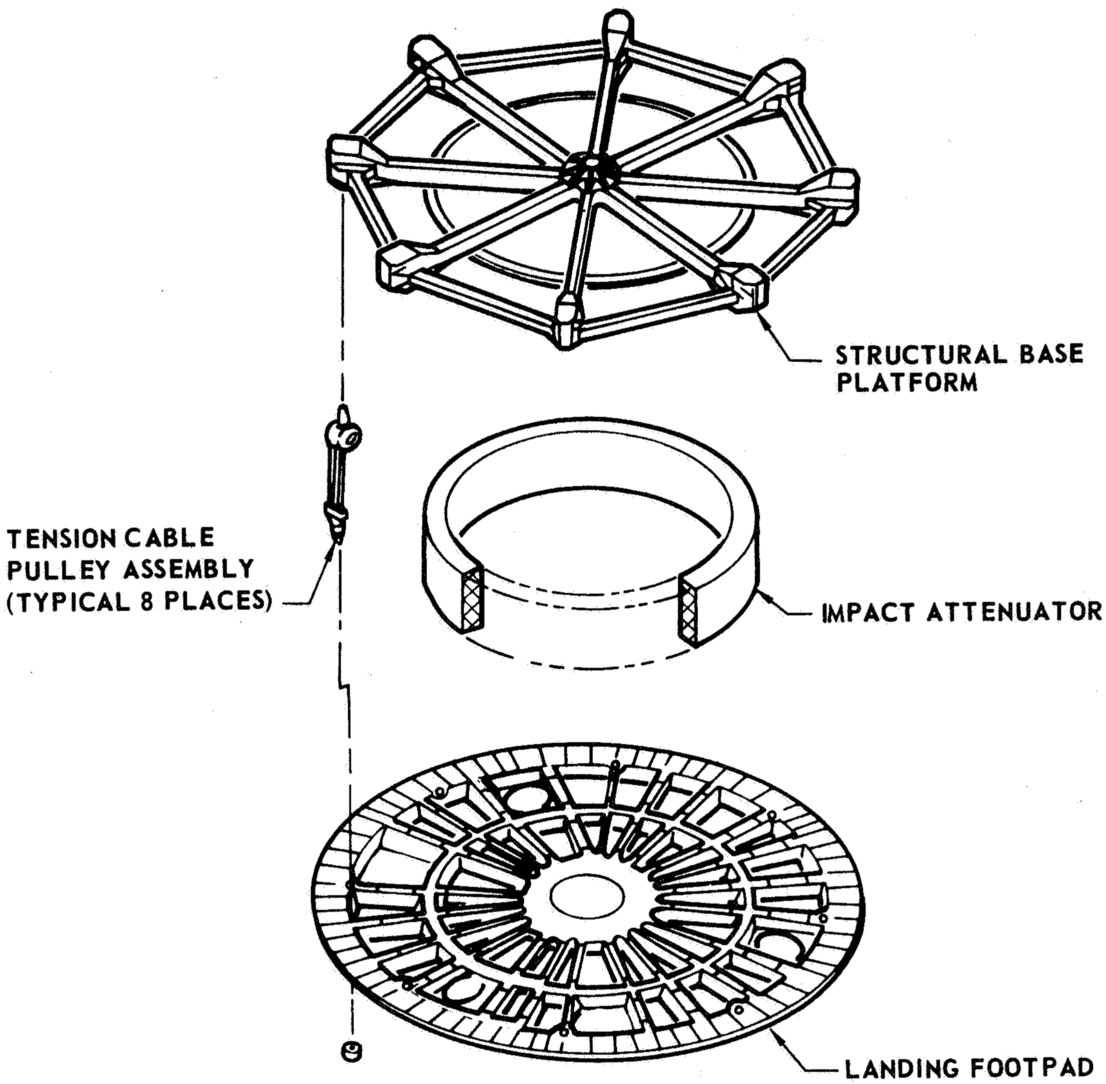


FIGURE 4.3.1–6

tie the footpad and base platform together, preventing separation during landing impact and forcing total attenuator loading.

The base platform is made up of eight equally spaced radial titanium I-beams and is the mounting base for the surface payload, and the capsule support equipment (See Figure 4.3.1-7).

The three terminal descent engines are mounted on the base platform, with the nozzles flush with the footpad. The nozzles are frangible and will be crushed during landing. Terminal propulsion system tankage is mounted on the base platform.

The solar panels are supported by the top of the surface payload and a continuous ring structure. The ring is supported by four bipods which are attached to the base platform.

The deorbit motor is supported by four struts that are attached to the ring structure at the bipod fittings by explosive bolts and straddle the surface payload. From the same structure under the deorbit motor on the vehicle centerline, the parachute catapult is suspended in a hole in the surface payload and bears on the base platform. The deorbit motor and the upper section of the support structure is jettisoned by explosive bolts and springs, after motor firing. The remaining portion of the structure, which supports the parachute installation, is separated by activating explosive bolts at the base of each tube.

The parachute riser assembly is attached to the base of the four tubes above the mounting plane. The lower support structure, parachute canister and the thermal curtain are removed by differential drag when the parachute is released.

LANDER GENERAL ARRANGEMENT

CONCEPT III

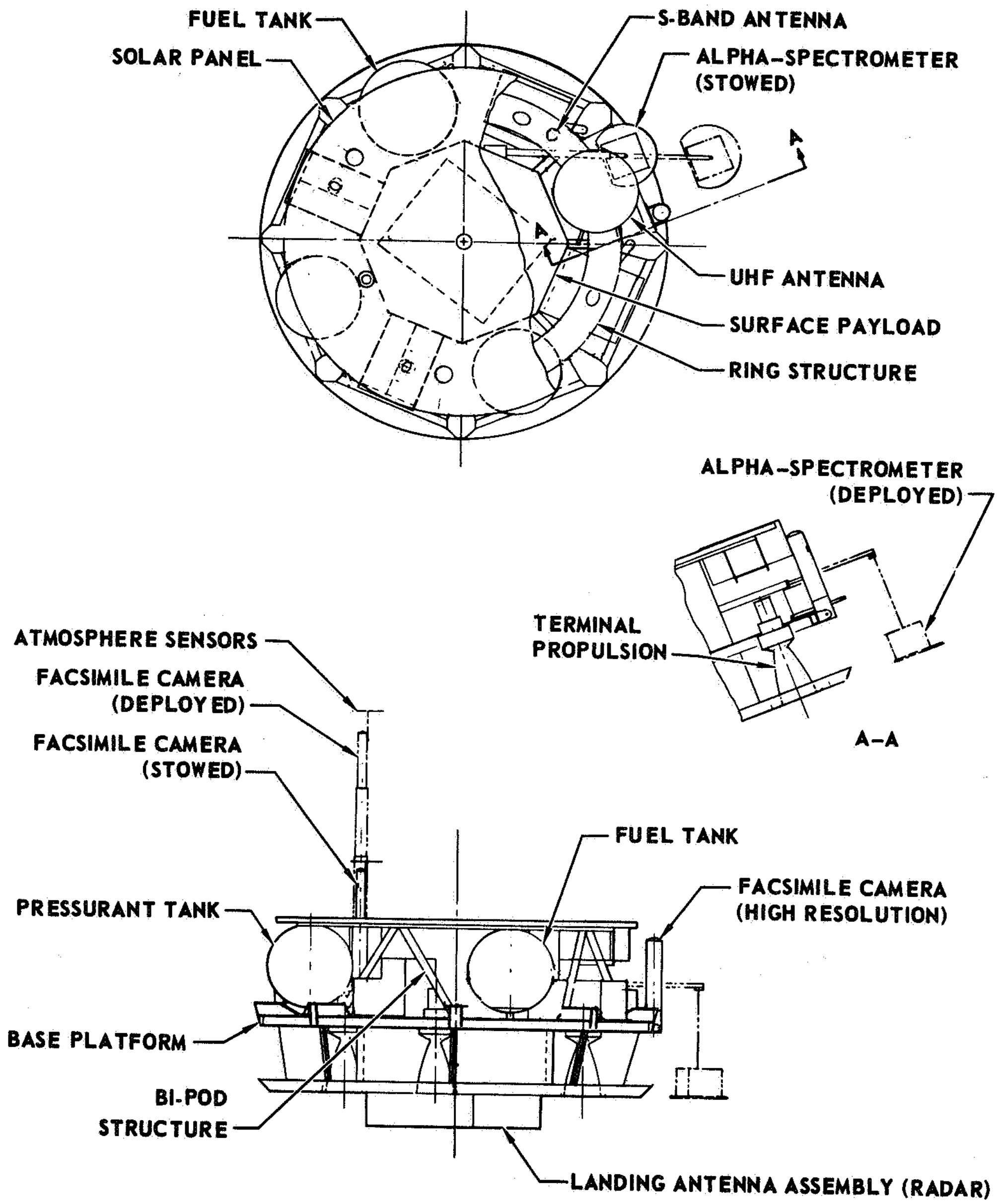


FIGURE 4.3.1-7

4.3.2 MAJOR SYSTEMS - The flight hardware elements of the major systems of the capsule are discussed in this section in the following order: science (both entry and landed), communication, power, guidance and control, sequencer, thermal control, propulsion, auxiliary aerodynamic decelerator, aeroshell, landing system, canister and adapter.

4.3.2.1 Science. - The science subsystem for the 90 day mission duration is nearly identical to the science subsystem for the 1 day mission; the only difference is in the operating procedure, electrical power utilized, the data generated, and the availability of commands.

Requirements and Constraints - The performance requirements in Figure 4.3.2.1-1 were obtained from the Mars Mission Objectives document, while the constraints were obtained from the nominal point design for the 90 day mission. The operating mean time before failure (greater than 50 000 hours) is only 5 times that for the one day mission (10 000 hours), even though the ratio of the mission durations is 90:1. This is due in part to the fact that a longer period is used between measurements for the long duration mission. The entry science requirements and constraints in Figure 4.3.2.1-2 are identical to those for the short duration mission since the entry sequences are identical.

Physical Description - The parameters for the science instruments in Figure 4.3.2.1-3 for the 90 day mission do not differ from those for the 1 day mission. There are some differences in the instruments for the longer duration missions; however, they are in characteristics other than those shown in this table. For example, for the alpha spectrometer, operation for a few days immediately following landing allows enough time for counting so that it is not necessary to operate it at night. Thus, insulation and electrical heaters are not required for the alpha spectrometer. The science equipment total weight is 53 lb.

Operational Description - The deployments which are carried out immediately following landing are illustrated in Figure 4.3.2.1-4. In order to transmit the wide angle facsimile camera data before the orbiter passes out of sight over the horizon, it is necessary to deploy this camera immediately and initiate scanning as soon as possible following landing.

SURFACE SCIENCE SUBSYSTEM REQUIREMENTS AND CONSTRAINTS

PERFORMANCE REQUIREMENTS

IMAGES

TO BE OBTAINED WITHOUT RELIANCE ON REGENERATED POWER

"LOW RESOLUTION"

RESOLUTION: 0.1°/LINE

COVERAGE: 4 SCENES LOCATED AT RIGHT ANGLES. SCENE SIZE OF 25° IN AZIMUTH AND 70° IN ELEVATION

COLOR: NO

"HIGH RESOLUTION"

RESOLUTION: 0.01°/LINE

COVERAGE: 4 SCENES LOCATED WITHIN "WIDE ANGLE" COVERAGE. SCENE SIZE OF 5° x 5°

COLOR: NO

THIS GIVES APPROXIMATELY 10^7 TOTAL BITS

METEOROLOGY

	ACCURACY
PRESSURE VARIATIONS WITH TIME	± 5 PERCENT OF VALUE AT KNOWN TIME 0.5 mb THRESHOLD
TEMPERATURE VARIATIONS WITH TIME	± 2 PERCENT OF VALUE AT KNOWN TIME 3°K THRESHOLD
WIND VELOCITY VARIATIONS WITH TIME	± 5 PERCENT OF VALUE AT KNOWN TIME 5 METERS/SEC THRESHOLD
WIND DIRECTION VARIATIONS WITH TIME	± 10° AT KNOWN TIME
SPECIFIC HUMIDITY (ATMOSPHERE) VARIATIONS WITH TIME	DEW/FROST PT OF ± 2°C

SOIL COMPOSITION

ASSUME TO BE SAME AS SURVEYOR α–SPECTROMETER EXPERIMENT

SCIENCE COMMAND CAPABILITY

NONE

DATA CONSTRAINTS

RELAY: LESS THAN 1.1×10^7 BITS/RELAY PERIOD

ELECTRICAL POWER

TOTAL POWER: LESS THAN 24 WATTS

VOLTAGE: 28 ± 4 VOLTS

WEIGHT

TOTAL WEIGHT: LESS THAN 30 LB

RELIABILITY

OPERATING MTBF: GREATER THAN 10 000 HR

FIGURE 4.3.2.1–1

ENTRY SCIENCE SUBSYSTEM REQUIREMENTS AND CONSTRAINTS

PERFORMANCE REQUIREMENTS

PRESSURE VARIATIONS WITH ALTITUDE

ALTITUDE RANGE: FROM 60 KM TO AS LOW AN ALTITUDE AS PRACTICAL

ACCURACY: ±5 PERCENT OF VALUE AT KNOWN ALTITUDE, THRESHOLD ACCURACY 0.5 MB

DENSITY VARIATIONS WITH ALTITUDE

ALTITUDE RANGE: FROM 60 KM TO AS LOW AN ALTITUDE AS PRACTICAL

ACCURACY: ±5 PERCENT OF VALUE AT KNOWN ALTITUDE, THRESHOLD ACCURACY 1×10^{-6} GM/CM^3

TEMPERATURE VARIATIONS WITH ALTITUDE

ALTITUDE RANGE: FROM 60 KM TO AS LOW AN ALTITUDE AS PRACTICAL

ACCURACY: ±2 PERCENT OF VALUE (°K) AT KNOWN ALTITUDE, THRESHOLD ACCURACY 3°K

COMPOSITION VARIATIONS WITH ALTITUDE (INCLUDING WATER VAPOR)

ALTITUDE RANGE: BELOW 50 KM

MASS NO. RANGE: 10–60 (REQUIRED DETECTION ON CO_2, H_2, Ar, AND H_2O)

ACCURACY: ±5 PERCENT IF GREATER THAN 50 PERCENT OF TOTAL
±10 PERCENT FOR OTHER CONSTITUENTS

DATA CONSTRAINTS

TOTAL DATA RATE: LESS THAN 350 BITS/SEC

ELECTRICAL POWER

TOTAL POWER: LESS THAN 16 WATTS

VOLTAGE: 24 ±4 VDC (DURING ENTRY NEAR BATTERY DEPLETION)
28 ±4 VDC (PRIOR TO ENTRY)

WEIGHT

TOTAL WEIGHT: LESS THAN 23 LB

RELIABILITY

OPERATING MTBF: GREATER THAN 10 000 HR

FIGURE 4.3.2.1–2

SCIENCE PAYLOADS

LANDED SCIENCE	WEIGHT LB	SIZE IN.	POWER WATTS	DIRECT DATA BITS/DAY	RELAY DATA BITS/DAY
CAMERAS	10	3 DIA x 18 1.5 DIA x 10	15	–	10^7
ATMOSPHERIC INSTRUMENTS	10		7	10^3	9.4×10^3
PRESSURE		2 DIA x 4 (2)			
TEMPERATURE		1 DIA x 1.9 (2)			
HUMIDITY		0.5 DIA x 2, 2 x 3 x 0.5			
WIND		6 DIA x 2			
ALPHA SPECTROMETER	10	6.75 x 5.25 x 4.6 6 x 6 x 6	2	–	8.4×10^3
MASS SPECTROMETER	(9)	(3 x 9 x 14)	(8)	–	(1.6×10^3)
TOTAL	30 LB		24 WATTS	10^3 BITS/DAY	17.8×10^3 BITS/DAY PLUS IMAGES

ENTRY SCIENCE	WEIGHT LB	SIZE IN.	POWER WATTS	AVERAGE DATA RATE BITS/SEC
MASS SPECTROMETER	9	3 x 9 x 14	8	80
ACCELEROMETER	2	2.75 x 1.75 x 2	4	150
PRESSURE SENSORS (6)	5	2 DIA x 2	3	48
TEMPERATURE SENSORS (2)	1	1 DIA x 1.9	0.02	16
HUMIDITY	1	0.5 DIA x 2, 2 x 3 x 0.5	0.02	8
MARGIN	5			
TOTAL	23 LB		15.04	302 BITS/SEC

FIGURE 4.3.2.1–3

VIEWING AND DEPLOYMENT CAPABILITY

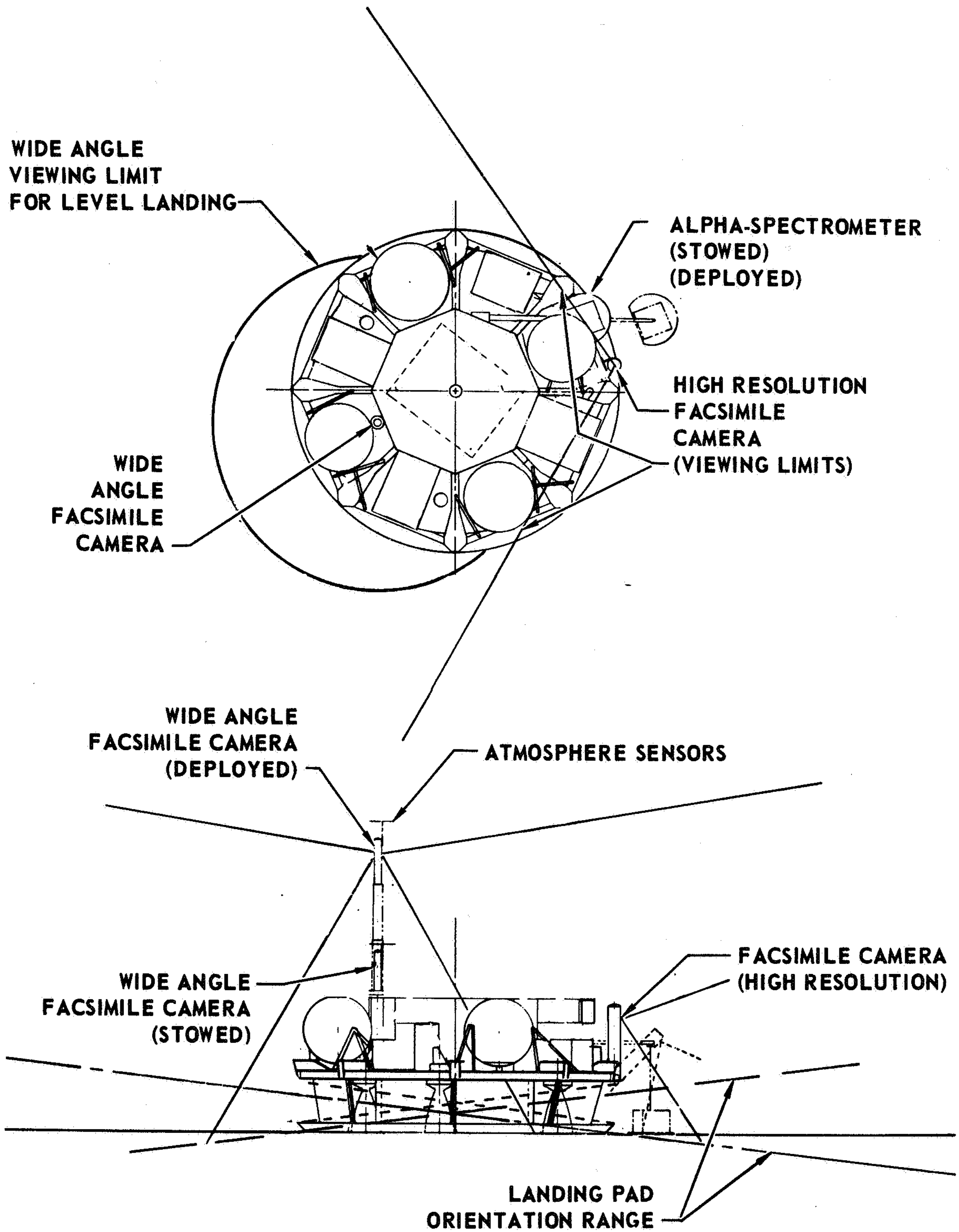


FIGURE 4.3.2.1-4

Since the facsimile camera operations are repeated several times during the first few days, as illustrated in Figures 4.3.2.1-5 and -6, it is possible to use the wide angle photographs to select the areas for the high resolution photographs and to determine the distance for focus of the high resolution camera.

The facsimile camera and the alpha spectrometer are not operated over most of the 90 day mission; however, measurements are made of the pressure, temperature, water vapor, and wind over the entire 90 days. In order to keep the data rate low, meteorological measurements, which are made every 15 minutes during the first few days, are made only once every 100 minutes for the remaining part of the mission. This produces a very small amount of data, approximately 10^3 bits per day, for most of the mission.

If a wind anemometer, which requires 4 watts, is used continuously, an extremely large amount of electrical energy would be required. For an instrument that uses this much power, it is necessary to operate it for only 1 minute in every 100 minutes in order to keep the total number of watt-hours down. It is possible to delete the S-band communications for 1 day and use this electrical energy to measure the wind continuously, to determine the maximum wind velocity, the time below the instrument wind velocity threshold, the average wind, and the instantaneous wind for 15 minute periods during the entire day. The wind data is stored and then transmitted on the next day, when the S-band communications system is again used. The development of an anemometer which uses much less than 4 watts would, however, be very desirable, to allow continuous wind measurements over the full 90 days.

It would also be desirable to have an orbiter period such that near the end of the 90 days it again becomes possible to have relay communications, so that the facsimile camera can be operated in order to determine if any seasonal changes have taken place over this period.

Development Status - All of the instruments are available, so the long duration mission can be carried out with a minimum of instrument development. If a wind anemometer can be developed that is sensitive enough to measure the low velocity winds at the low surface pressures on Mars and that requires only a small amount of electrical power, then the wind measurements will be greatly improved.

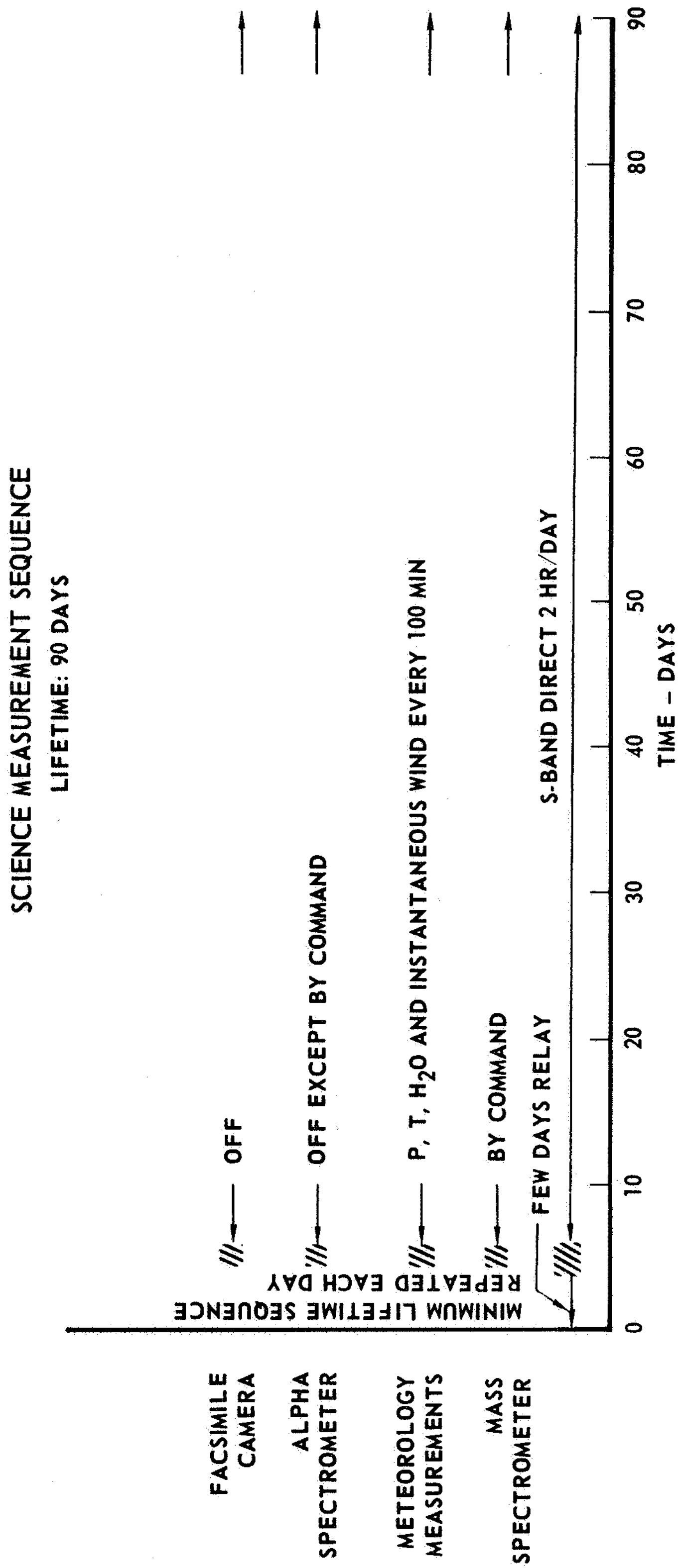


FIGURE 4.3.2.1–5

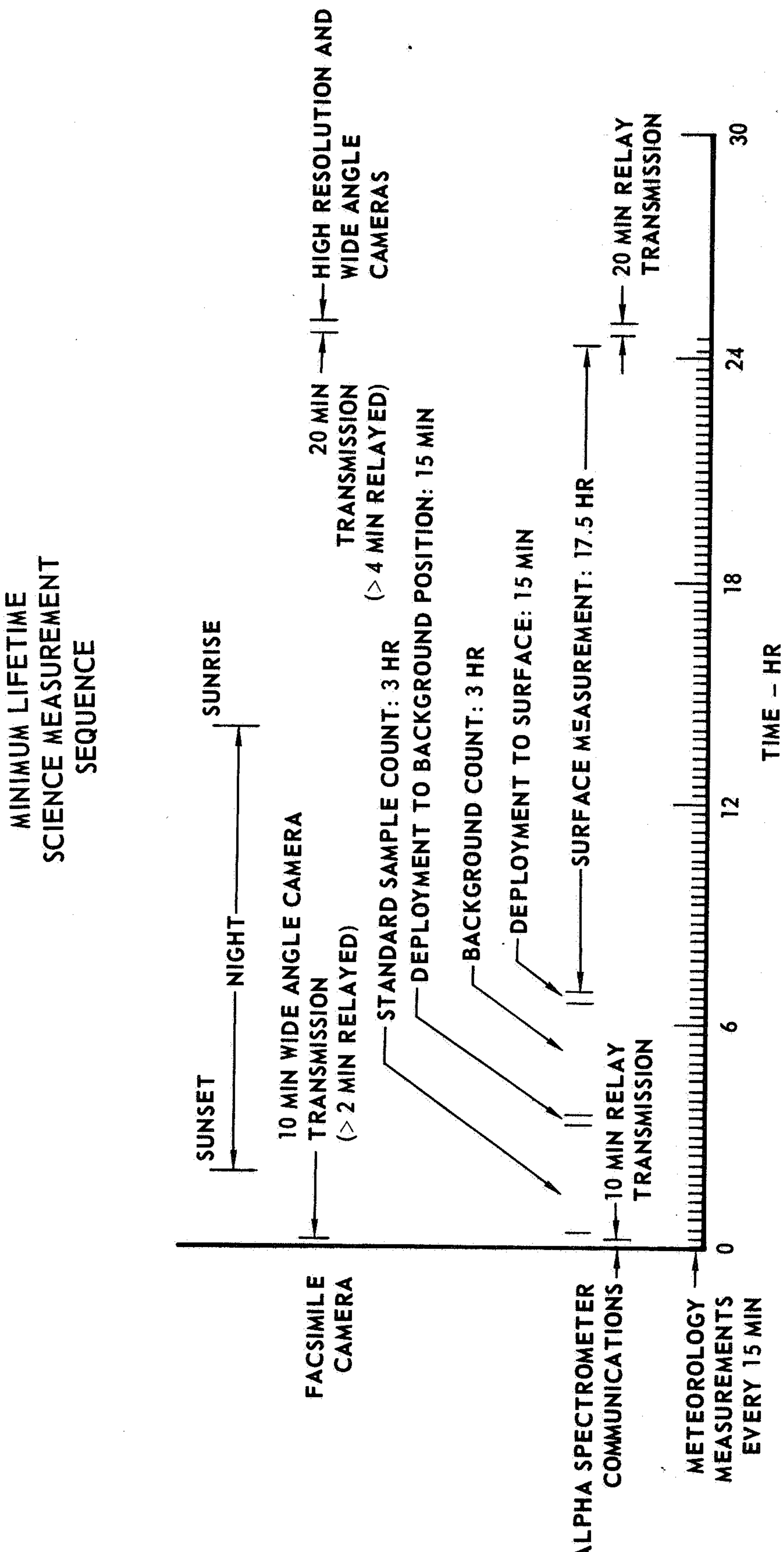


FIGURE 4.3.2.1-6

4.3.2.2 Communications system functional description. - The function of the communications equipment is to collect the scientific and engineering data, format the data sources into a coherent data stream, temporarily store the information, and transmit the data at the appropriate times. Table 4.3.2.2-1 lists the capsule data requirements.

The functional description is divided into two parts; the capsule-mounted communications equipment which weighs 74.7 lb and the orbiter-mounted communications support equipment weighing 37.4 lb.

4.3.2.2.1 Capsule mounted communications equipment: The capsule mounted communications equipment contains the following systems:

a) Entry antenna

b) Entry radio

c) Entry telemetry

d) Entry instrumentation

e) Relay antenna

f) Relay radio

g) Landed storage

h) Landed telemetry

i) Landed instrumentation

j) Direct antenna

k) Direct radio

l) Direct command

Figure 4.3.2.2-1 is a block diagram of the equipment, and Table 4.3.2.2-2 is a list of the salient characteristics of the design. Table 4.3.2.2-3 is a list of the physical parameters of the equipment.

System Description - The equipment has five operating modes in addition to the in flight checkout and memory dump modes. These modes are:

a) Deoribt descent

TABLE 4.3.2.2-1

TOTAL NUMBER OF PARAMETERS

SUBSYSTEMS	DEORBIT	ENTRY	TERMINAL DESCENT	LANDING RELAY	S-BAND
UHF TRANSMITTER	8	8	8	8	–
S-BAND TRANSMITTER	–	–	–	–	8
TELEMETRY	19	19	19	19	19
THERMAL CONTROL	62	62	56	16	16
AERODYNAMIC	7	17	18	–	–
RADAR ALTIMETER	2	28	28	–	–
LANDING RADAR	2	18	40	–	–
GUIDANCE & CONTROL	44	44	44	–	–
SURFACE SEQUENCER & TIMER	4	4	4	5	5
POWER	30	30	30	25	5
PROPULSION	22	13	8	3	–
PYROTECHNICS	10	13	14	–	–
MASS SPECTROMETER – SCIENCE	–	9	9	–	–
ALPHA – SPECTROMETER – SCIENCE	–	–	–	5	–
FACSIMILE CAMERAS – SCIENCE	–	–	–	18	–
ACCELEROMETERS – SCIENCE	–	4	4	–	–
WEATHER DATA – SCIENCE	–	–	–	8	8
ATMOSPHERE DATA – SCIENCE	–	9	9	9	9
	210	278	291	116	70

CAPSULE MOUNTED COMMUNICATIONS EQUIPMENT

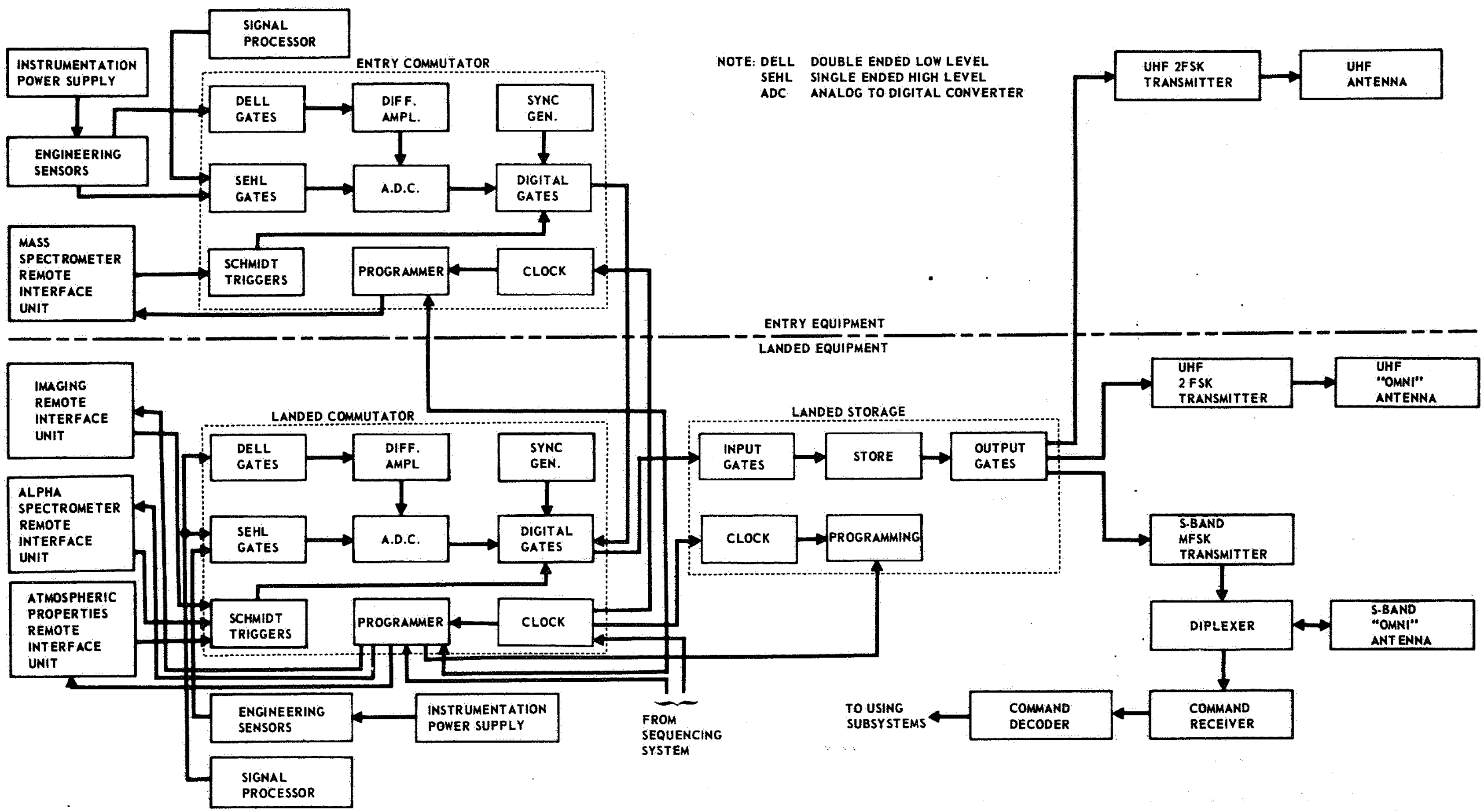


FIGURE 4.3.2.2-1

TABLE 4.3.2.2-2

COMMUNICATIONS CHARACTERISTICS SUMMARY TABLE

ENTRY	• FREQUENCY = 400 MHz BAND • TRANSMITTER = 3 WATTS • TELEMETRY CHANNELS = 401 • DATA RATES = 440, 1320 BPS • STORAGE = 150 AND 50 SEC = 198,656 BITS • RECEIVING ANTENNA BEAMWIDTH = 120° • TRANSMITTING ANTENNA BEAMWIDTH = 90° • MODULATION = 2 FSK
LANDED RELAY	• FREQUENCY = 400 MHz BAND • TRANSMITTER = 5 WATTS • TELEMETRY CHANNELS = 116 • DATA RATE = 41.6 K BPS • STORAGE = 24.6 HR = 167 936 BITS • RECEIVING ANTENNA BEAMWIDTH = 120° • TRANSMITTING ANTENNA BEAMWIDTH = 120° • MODULATION = 2 FSK
LANDED DIRECT TELEMETRY	• FREQUENCY = 2300 MHz BAND • TRANSMITTER = 20 WATTS • TELEMETRY CHANNELS = 70 • DATA RATE = 1.088 BPS • STORAGE = 24.6 HR = 8192 BITS • RECEIVING ANTENNA BEAMWIDTH = 0.15° • TRANSMITTING ANTENNA BEAMWIDTH = 120° • MODULATION = MFSK
LANDED DIRECT COMMAND	• FREQUENCY = 2300 MHz BAND • TRANSMITTER = 100 K WATTS • COMMAND RATE = 1 BPS • RECEIVING ANTENNA BEAMWIDTH = 120° • TRANSMITTING ANTENNA BEAMWIDTH = 0.15° • MODULATION = PSK-BCH

TABLE 4.3.2.2–3

CAPSULE COMMUNICATIONS EQUIPMENT

ITEM	SIZE (CU IN.)	WEIGHT (LB)	POWER – WATTS				
			DEORBIT	ATMOSP ENTRY	TERMINAL DESCENT	UHF RELAY	S-BAND DIRECT
ENTRY ANTENNA	13.5 DIA	3	–	–	–	–	–
ENTRY TRANSMITTER	40.9	3.9	15.3	15.3	15.3	–	–
ENTRY COMMUTATOR	297.7	11.9	2.9	3.7	3.7	–	–
ENTRY SIGNAL PROCESSOR	75.4	2.2	3.6	3.6	3.6	–	–
ENTRY ENGINEERING SENSORS	67.5	6.1	–	–	–	–	–
ENTRY MASS SPECTROMETER RIU	21	.8	–	.3	.3	–	–
RELAY ANTENNA	13.5 DIA	3	–	–	–	–	–
RELAY TRANSMITTER	51.4	4.9	–	–	–	23.4	–
LANDED CORE STORAGE	283	10.25	–	2.3	2.3	.3	.014
LANDED COMMUTATOR	106.2	4.3	.97	1.24	1.24	.4	.2
LANDED SIGNAL PROCESSOR	17.4	.5	.83	.83	.83	.8	.83
LANDED ENGINEERING SENSORS	9.6	.54	–	–	–	–	–
IMAGING RIU	45	1.8	–	–	–	1.3	–
ALPHA SPECTROMETER RIU	20	.8	–	–	–	.5	–
ATMOSP PROPERTIES RIU	22	.9	.2	.3	.3	.08	.06
S-BAND ANTENNA	2.3 DIA	.3	–	–	–	–	–
S-BAND DIPLEXER	36	1.2	–	–	–	–	–
S-BAND TWTA	103.9	5.5	–	–	–	–	73
S-BAND FREQ SYNTHESIZER	97.5	4.5	–	–	–	–	2.8
COMMAND RECEIVER	40	2.3	–	–	–	–	2.5
COMMAND DECODER	43.7	1.1	–	–	–	–	.2
S-BAND EXCITER	40	4	–	–	–	–	8.9
S-BAND OUTPUT FILTER	25	0.8	–	–	–	–	–

NOTE: R.I.U. = REMOTE INTERFACE UNIT

b) Atmospheric entry

c) Terminal descent

d) UHF relay

e) S-Band direct

o Deorbit Descent Mode - The deorbit descent mode is used from separation to the initiation of atmospheric entry. The 210 data channels are formatted by the combined entry telemetry commutator and landed telemetry commutator into an interlaced tube format at a 440 bps rate. This bitstream is subbit encoded to 3960 bps and transmitted to the spacecraft by the 3 watt non-coherent frequency shift keyed (FSK) entry transmitter. Subbit encoding is used here to keep a constant 3960 bps transmission rate thereby avoiding mode switching in the spacecraft bit synchronizer. Delay storage is not required during descent.

o Atmospheric Entry Mode - The atmospheric entry mode begins at 800 000 feet and continues until the initiation of terminal descent. During atmospheric entry 278 data channels are formatted by the combined entry and landed telemetry commutators into an interlaced tube format at 1320 bps. This bitstream is transmitted in real time and also transmitted twice in delayed time to assure that the data gathered during the plasma "blackout" is reliably transmitted to the orbiter at least one time. The core storage unit in this mode is programmed as a tapped delay line. The length of the line is 150 seconds, and the tap is at 50 seconds. The real time plus the two delayed time bitstreams are combined into a 3960 bps stream, and transmitted to the orbiter over the entry UHF transmitter.

o Terminal Descent Mode - The terminal descent mode begins when the landing radar comes on and lasts until landing. The system operation during terminal descent is the same as that in atmospheric entry except that a different set of data is collected.

The data formats for the preceding three modes are shown in Figure 4.3.2.2-2.

ENTRY DATA SYSTEM FORMATS

DEORBIT DESCENT MODE

5	1	.5	.1	.05	.01 SPS
2 CH INC 1 SPARE	23 CH INC 3 PF SYNC +1 ID +10 SPARES	21 CH INC 1 ID +4 SPARES	95 CH INC 1 ID +13 SPARES	20 CH INC 1 ID +8 SPARES	100 CH INC 10 SPARES

261 CHANNELS
55 WPS
440 BPS
11 WPM_iF
5500 WPM_aF

NOTE: PF = PRIME FRAME
ID = IDENTIFICATION WORD

ATMOSPHERIC ENTRY MODE

5	1	.5	.1	.05	.01 SPS
4 CH	113 CH INC 3 PF SYNC +1 ID +11 SPARES	40 CH INC 1 ID +7 SPARES	100 CH INC 1 ID +4 SPARES	20 CH INC 1 ID +8 SPARES	100 CH INC 25 SPARES

377 CHANNELS
165 WPS
1320 BPS
33 WPM_iF
16 500 WPM_aF

TERMINAL DESCENT MODE

5	1	.5	.1	.05	.01 SPS
4 CH	124 CH INC 3 PF SYNC +1 ID	19 CH INC 1 ID +2 SPARES	97 CH INC 1 ID +4 SPARES	20 CH INC 1 ID +8 SPARES	80 CH INC 11 SPARES

344 CHANNELS
165 WPS
1320 BPS
33 WPM_iF
16 500 WPM_aF

FIGURE 4.3.2.2–2

The entry transmitter power level was selected on the basis of the maximum power required in entry at a 5 deg central angle between the capsule and the orbiter at the time of entry. From Figure 3.2.2-7, for the worst case atmosphere, trajectory, and surface roughness, this is 25.4 dBm. This power requirement is not exceeded when parachute oscillations less than 30 deg occur in any atmosphere. Since the 25.4 dBm computation was made on a 2560 bps rate, 1.9 dB is added to account for the 3960 bps rate, bringing the required level to 27.3 dBm. In addition, 6.4 dB must be added for a 120 deg orbiter antenna over a 55 deg antenna. A 3 watt transmitter thus has a 1.1 dB excess margin. Referring to Table 3.2.2-2, this power level is equivalent to a 5060 km design range for the no multipath condition. Table 4.3.2.2-4 defines the link characteristics.

o Landed Relay Mode - The UHF relay mode consists of transmitting the stored diurnal cycle data plus real time imaging during periapse passage. The scientific and engineering data is collected at a sample rate of once each 15 minutes and stored in core. This data is interleaved with the imaging data, taken during the orbiter overflight, and relayed via a 400 MHz link.

The worst case landing dispersion due to atmospheric and entry uncertainties is ± 4.7 deg, i.e., a maximum spacecraft lead of 2.4 deg for a VM-9, -15 deg entry, and a maximum spacecraft lag of 7 deg for a VM-8, -15 deg entry (for a consistent 5 deg central angle at entry). From Figure 3.2.2-26 the minimal transmitter output power to transmit 10^7 bits is 2.5 watts for 4 minutes, or a 41.6 K bps rate. An additional .87 dB is required in order to use a 120 deg rather than a 90 deg orbiter antenna. Thus, the "required" power is 3.07 watts. From Figure 3.2.2-23, the maximum slope uncertainty at this rate is ± 3.2 minutes. The net result of these three factors, (atmospheric uncertainty, power/time for 10^7 bits, and slope uncertainty) is that 17.27 minutes of transmission time is required to assure the 4 minute, 10^7 bit "window." With 2.13 dB excess margin, the resultant power requirements is 5 watts, which corresponds (reference Table 3.2.2-2) to a design transmission range of 1480 km with a 120 deg orbiter antenna. Table 4.3.2.2-5 defines the link characteristics.

TABLE 4.3.2.2-4

ENTRY RELAY LINK TABLE

PARAMETER	VALUE	TOLERANCE
TRANSMITTER POWER, (3W)	34.8 dBm	+ .5 −.5
TRANSMITTING CIRCUIT LOSS	−.5 dB	+ .2 −.3
TRANSMITTING ANTENNA GAIN	3.2 dB	+ .5 −.5
SPACE LOSS (f = 400 MHz, R = 1347 Km)	−147.1 dB	+ .0 −.0
POLARIZATION LOSS	0.0 dB	+ .0 −.5
RECEIVING ANTENNA GAIN	2.7 dB	+ .0 −3.0
RECEIVING CIRCUIT LOSS	−1.0 dB	+ .2 −.2
NET CIRCUIT LOSS	−142.7 dB	+ .9 −4.5
TOTAL RECEIVED POWER	−107.9 dBm	+1.4 −5.0
RECEIVER NOISE SPECTRAL DENSITY (555°K)	−171.2 dBm/Hz	+ .2 −.7
BIT RATE (3960 PS)	36.0 dB-bps	+ .0 −.0
REQUIRED E/N_o	21.0 dB	+ .0 −.0
THRESHOLD POWER	−114.2 dBm	+ .2 −.7
MARGIN	6.3 dB	+2.1 −5.2

NOTES: 1) E/N_o INCLUDES 2.0 dB FOR DEGRADATIONS DUE TO BT > 1 AND THE BIT SYNCHRONIZER.

2) TRANSMITTING ANTENNA GAIN, SPACE LOSS AND E/N_o CORRESPOND TO THE WORST CASE COMBINATION OF THE TRAJECTORIES INVESTIGATED.

TABLE 4.3.2.2-5

LANDED RELAY LINK TABLE

PARAMETER	VALUE	TOLERANCE
TRANSMITTER POWER (5W)	37.0 dBm	+.5 −.5
TRANSMITTING CIRCUIT LOSS	−.5 dB	+.2 −.3
TRANSMITTING ANTENNA GAIN	1.4 dB	+.5 −.5
SPACE LOSS (f 400 MHz, R 1078 Km)	−145.1 dB	+.0 −.0
POLARIZATION LOSS	0.0 dB	+.0 −.5
RECEIVING ANTENNA GAIN	2.2 dB	+.0 −3.0
RECEIVING CIRCUIT LOSS	−1.0 dB	+.2 −.2
NET CIRCUIT LOSS	−143.0 dB	+.9 −4.5
TOTAL RECEIVED POWER	−106.0 dBm	+1.4 −5.0
RECEIVER NOISE SPECTRAL DENSITY (555°K)	−171.2 dBm/Hz	+.2 −.7
BIT RATE (41.6 kbps)	46.2 dB−bps	+.0 −.0
REQUIRED E/N_o	11.7 dB	+.0 −.0
THRESHOLD POWER	−113.3 dBm	+.2 −.7
MARGIN	7.3 dB	+2.1 −5.2

NOTES: 1) E/N_o INCLUDES 0.8 dB DEGRADATION DUE TO THE BIT SYNCHRONIZER
2) TRANSMISSION NEAR PERIAPSIS FOR 4 MIN PER PASS

Because of this choice of capsule-orbiter relative geometry at entry (5 deg central angle), the orbiter will lead the capsule by 2.4 deg at landing in the worst case (VM-9, -15 deg entry). Allowing for 20 seconds warmup time, and transmitting at a power level (5 watts) and bit rate (41.6 k bps) commensurate with a 10^7 bit transmission on the second day permits 155 seconds of transmission and return of 6.45 M bits after landing.

o Direct Mode - The direct to Earth link is a S-band 16 tone frequency shift keyed link using a 20 watt travelling wave tube amplifier (TWTA). With a nominal 24 May 1974 landing, from Figure 3.2.2-18, a 2 bps rate can be supported. The generic link table for this system is given in Table 3.2.2-4. However, this rate slowly decreases to .7302 bps by 22 August 1974, which is below the minimum .75 bps rate. Given that the power system can support the TWTA for only 2 hours, the required 70 channels (544 bits) can be sampled at an average rate of only once every 108.8 minutes. Rounding this sampling rate offf to once every 100 minutes, the mean collection rate is .09066 bps, and the dump rate is 1.088 bps for a total daily bit collection of 7833.6 bits. Table 4.3.2.2-6 is a link table of this system for 22 August 1974. The transmission for only 2 hours, as opposed to the optimal 6 (for maximum bit transfer) gives a slightly higher allowable data rate, resulting in a 0.5 dB excess system margin.

o Command - An extended mission of 90 days will require a command system. Figure 3.2.2-22 shows that, for the command link defined in Table 3.2.2-6, a 10 kw transmitter can support the mission until 1 June 1974, while a 100 kw transmitter has adequate excess margin beyond 1 August 1974. Table 4.3.2.2-7 gives the command link margin for a 100 kw transmitter for 22 August 1974.

4.3.2.2.2 Orbiter-mounted support communications equipment: The spacecraft mounted support communications equipment contains the following subsystems:

a) UHF antenna

b) UHF radio

c) Storage

d) Telemetry

e) Command

TABLE 4.3.2.2-6

DIRECT LINK TABLE

- MFSK
- 24 MAY 1974 + 90 DAYS = 22 AUG 1974
- SUB EARTH LATITUDE = 24°N

PARAMETER	VALUE	TOLERANCE
TRANSMITTER POWER (20W)	43.0 dBm	+ .5 –.5
TRANSMITTING CIRCUIT LOSS	–.7 dB	+ .1 –.1
TRANSMITTING ANTENNA GAIN	–.2 dB	+1.0 –1.0
SPACE LOSS (f = 2295 MHz, R = 388 x 10^6 KM)	–271.4 dB	+ .0 –.0
POLARIZATION LOSS	–.1 dB	+ .1 –.6
RECEIVING ANTENNA GAIN	+61.0 dB	+1.0 –1.0
RECEIVING ANTENNA POINTING LOSS	–.1 dB	+ .1 –.1
RECEIVING CIRCUIT LOSS	–.1 dB	+ .1 –.1
NET CIRCUIT LOSS	–211.6 dB	+2.4 –2.9
TOTAL RECEIVED POWER	–168.6 dBm	+2.9–3.4
RECEIVER NOISE SPECTRAL DENSITY (45°K ± 10°K)	–182.1 dBm/Hz	+ .9 –1.1
REQUIRED S/N (1.088 bps)	8.7 dB-Hz	+ .0 –1.1
THRESHOLD POWER	–173.4 dBm	+ .9 –2.2
MARGIN	4.8 dB	+5.1–4.3

XAG ASSUMES 34° SOUTH SLOPE

TABLE 4.3.2.2-7

COMMAND LINK TABLE

PARAMETER	VALUE	TOLERANCE
TRANSMITTER POWER (100 KW)	80.0 dBm	+ .5 −.5
TRANSMITTING CIRCUIT LOSS	−.4 dB	+ .1 −.1
TRANSMITTING ANTENNA GAIN	60.0 dB	+ .8 −.8
TRANSMITTING ANTENNA POINTING LOSS	−.1 dB	+ .0 −.1
SPACE LOSS (2113 MHz, 388 x 10^6 Km)	−270.7 dB	+ .0 −.0
POLARIZATION LOSS	−.1 dB	+ .1 −.6
RECEIVING ANTENNA GAIN	−.2 dB	+1.0 −1.0
RECEIVING CIRCUIT LOSS	−1.3 dB	+ .2 −.2
NET CIRCUIT LOSS	−212.8 dB	+2.2 −2.8
TOTAL RECEIVED POWER	−132.8 dBm	+2.7 −3.3
RECEIVER NOISE SPECTRAL DENSITY (644°K)	−170.5 dBm/Hz	+ .8 −.4
CARRIER MODULATION LOSS	−2.4 dB	+ .3 −.3
RECEIVED CARRIER POWER	−135.2 dBm	+3.0 −3.6
CARRIER APC NOISE BANDWIDTH (2 B_{LO} = 20 Hz)	13.0 dB−Hz	+ .0 −1.0
CARRIER PERFORMANCE – COMMAND DETECTION		
THRESHOLD SNR IN 2 B_{LO}	8.0 dB	+1.0 −1.0
THRESHOLD CARRIER POWER	−149.5 dBm	+1.8 −2.4
PERFORMANCE MARGIN	14.3 dB	+5.4 −5.4
COMMAND CHANNEL PERFORMANCE		
MODULATION LOSS	−8.5 dB	+ .2 −.2
RECEIVED COMMAND SUBCARRIER POWER	−141.3 dBm	+2.9 −3.5
BIT RATE (1 bps)	0.0 dB−bps	+ .0 −.0
REQUIRED E/N_o	15.7 dB	+1.0 −1.0
THRESHOLD SUBCARRIER POWER	−154.8 dBm	+1.8 −1.4
PERFORMANCE MARGIN	13.5 dB	+4.3 −5.3
SYNCHRONIZATION CHANNEL PERFORMANCE		
MODULATION LOSS	−5.5 dB	+ .2 −.3
RECEIVED SYNC SUBCARRIER POWER	−138.3 dBm	+2.9 −3.6
SYNC APC NOISE BANDWIDTH (2 B_{LO} = 2 Hz)	3.0 dB−Hz	+ .0 −.0
THRESHOLD SNR IN 2 B_{LO}	15.7 dB	+1.0 −1.0
THRESHOLD SUBCARRIER POWER	−151.8 dBm	+1.8 −1.4
PERFORMANCE MARGIN	13.5 dB	+4.3 −5.4

Figure 4.3.2.2-3 is a block diagram of the equipment, and Table 4.3.2.2-8 is a list of the physical characteristics.

The UHF antenna receives both the entry and the landed signals. It is a helix with a gain of 3.22 dB and a beamwidth of 120 degrees. The diplexer transfers the entry telemetry signal at 3960 bps to the entry receiver, and the landed telemetry signal at 41.6 k bps to the landed relay receiver.

The entry receiver is a conventional double super-heterodyne frequency shift keyed receiver with a noise temperature of 555°K. The mark and space filters have a bandwidth of 10 kHz to accommodate a maximum 6 kHz Doppler, and thus operate at a bandwidth pulsewidth produce of 2-1/2. The entry bit synchronizer is a standard phase lock loop, integrate and dump device which incorporates a phase gate to track the direct path phasor and lock out the indirect (multipath) phasor. The entry bit synchronizer runs at a constant 3960 bps rate, independent of the entry telemetry system mode.

The landed relay receiver is similar to the entry receiver, but with 41.6 kHz bandwidth mark and space filters, and it operates at a bandwidth pulsewidth product of unity since no appreciable Doppler will be present. The landed realy bit synchronizer is a standard phase lock loop, integrate and dump device. The special phase gate is not required for this receiver, because the landed system only transmits at high evelation angles, where multipath is not a problem. The synchronizer rate is 41.6 bps.

Both of the receivers feed the data distribution unit, which in turn either routes the incoming data directly to the orbiter or to the orbiter mounted support tape recorder. The data distribution unit is the central interface equipment between the orbiter and the capsule (prior to separation) and the capsule support equipment. This central data interface simplifies the capsule-orbiter interface problems. In addition to routing the entry and relay data to the orbiter it also routes the radio commands, via the orbiter command receiver, to the support equipment command decoder and routes the support equipment commutator and the payload commutators (prior

SPACECRAFT MOUNTED
SUPPORT COMMUNICATIONS EQUIPMENT

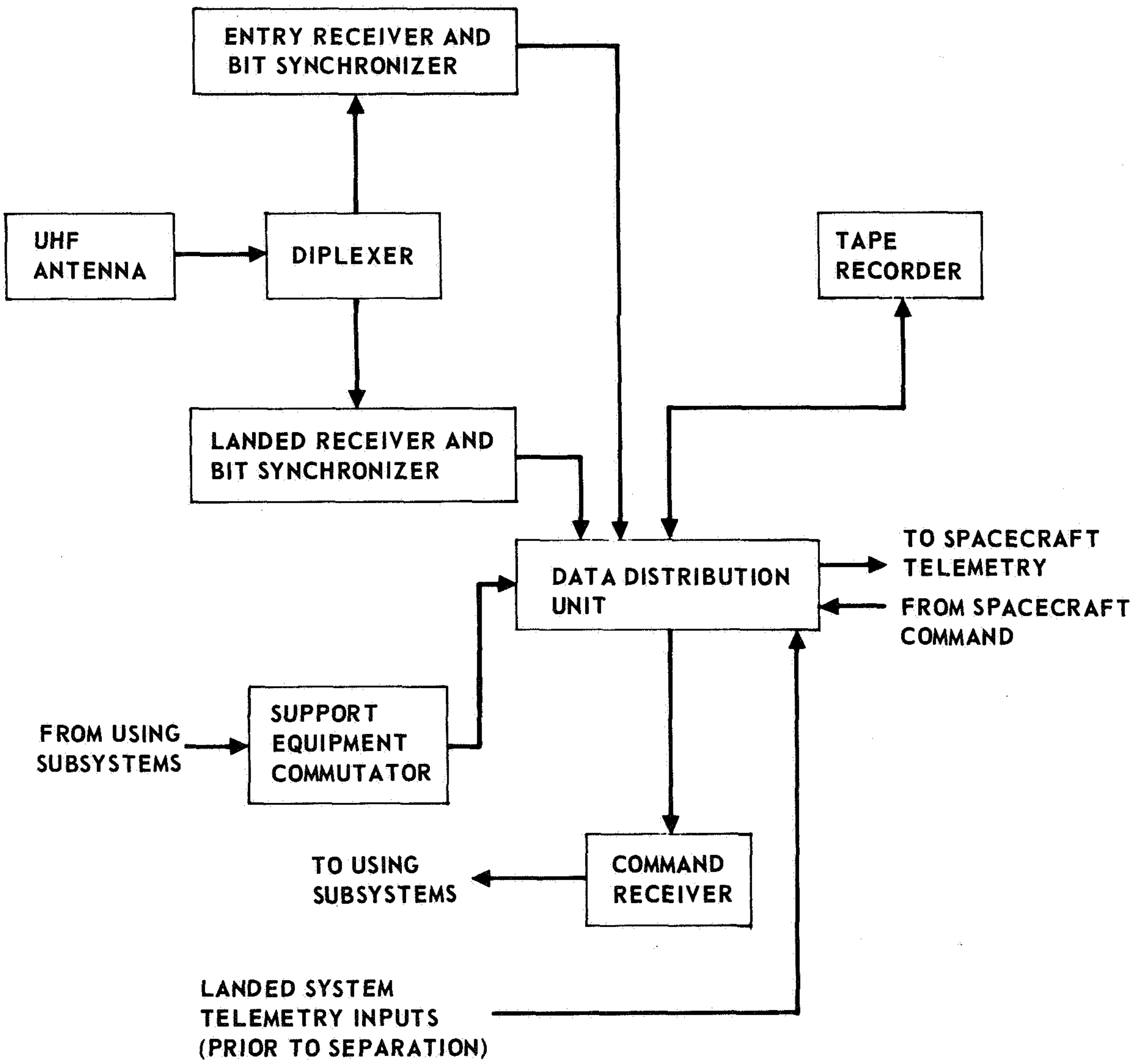


FIGURE 4.3.2.2-3

TABLE 4.3.2.2–8

SPACECRAFT MOUNTED COMMUNICATIONS EQUIPMENT

ITEM	SIZE (CU IN.)	WEIGHT (LB)	POWER (W)
UHF ANTENNA	DIA = 9.4, L = 11.2	0.2	–
DIPLEXER	8	.8	–
UHF RECEIVERS	240	11.4	5.8
TAPE RECORDER	414	13.8	2.0
DATA DISTRIBUTION UNIT	128	3.2	2.0
SUPPORT COMMUTATOR	64	2.6	2.0
COMMAND DECODER	168.5	5.4	1.3

to separation) to the orbiter telemetry system.

The tape recorder serves as an intermediate storage device between the entry and landed relay transmissions and the spacecraft retransmission. At moderate bit packing densities of 1 k bpi per track, 257 feet of one inch tape are required to assure the transmission of a 10^7 relay picture with the net landing time and surface slope uncertainties. This recorder could accommodate a 3.03 hour deorbit/entry time. The entry time range, however, is up to 9 hours. Thus, a 764 feet tape is required. With the "nominal" 3.42 hour orbital descent time only 290 feet of tape would be required. The recorder may not be required in the support equipment area if the orbiter recorder can be made available.

The support equipment commutator serves as a remote multiplexer to the spacecraft telemetry. This commutator monitors the operation of the two UHF receivers, the bit synchronizers, the tape recorder, the data distribution unit, the command decoder, and the in flight checkout equipment.

The spacecraft command receiver receives all the commands from Earth and strips all of the commands to the support equipment and the capsule prior to separation to the support equipment command decoder. The support equipment decoder then routes the commands to the using subsystems.

4.3.2.2.3 Development status: No special development problems are anticipated for any of the equipment in the conceptual designs. Every type of component required in the designs has been successfully flown on previous missile and spacecraft programs.

A sterilizable tape recorder, not required in the conceptual designs, would require development. Although some work as progressed on "high temperature" tapes, and various recorder components, no comprehensive "tape system" work has been undertaken. Such an effort would be principally directed toward the definition of the long term chemical effects of the potentially damaging sterilization process. A sterilizable tape recorder could probably be developed in 1-1/2 to 2 years. However, some uncertainty would remain as to its reliability after an extended life.

4.3.2.3 Power system. - The electrical power system provides the basic energy storage and power management for the lander and surface payload electrical equipment. Concept III assumptions pertinent to the power system are:

a) 90-day mission goal

b) solar cell array regenerative power source

c) 10°N latitude landing site

d) 34° maximum slope inclination

e) 24 May 1974 landing date

f) isotope heat for surface payload

g) electrical energy contingency for thermal control during one continuous -190°F day

h) relay link communication initially

i) direct communication later

j) one diurnal cycle capability without regenerative power source.

4.3.2.3.1 Design requirements and goals: To provide electrical power for a 90-day mission, a regenerative power source (solar cell array) is required. In addition, energy storage is required to provide power for descent and entry, power for one diurnal cycle without any contribution from the solar array, power during night periods, and power for thermal control during -190°F conditions.

Trade studies have shown, in general, that a lighter weight landed system evolves when the total required energy storage is divided into two components: that for descent and entry, and that for surface operation. This weight saving occurs because the descent/entry battery may be allowed to freeze after touchdown, while surface operations require a battery to survive the cold, night temperatures. Hence, a thermal control weight penalty is involved which is dependent upon the mass and surface area of the battery to be protected. Generally, the added thermal control weight for a single, compact battery, sized for descent and entry plus surface operation, is greater than the weight penalty of reduced energy density for two, separate batteries: one for descent and entry, and one for surface operation.

The electrical power and energy requirement for descent and entry are graphically illustrated by the power profile for Figure 4.3.2.3.1-1. For a descent period of 3.5 hours, 400 watt-hours arerequired for equipment operation. Allowing a 6% distribution loss factor, the required battery capacity is 425 watt-hours.

The electrical power and energy requirements for the initial diurnal cycle of surface payload operations are graphically illustrated by the power profile of Figure 4.3.2.3.1-2. Isotope heat is provided for normal surface payload thermal control, but an electrical energy contingency is required, as indicated by the shaded area, to provide the thermal control power for one day of continuous -190°F environment. The total requirement for the first day is 1175 watt-hours. Allowing a 6% distribution loss factor and assuming no solar array contribution, the required battery capacity is 1245 watt-hours.

The electrical power and energy requirements for each successive diurnal cycle, utilizing the relay mode of communication, are graphically illustrated by the power profile of Figure 4.3.2.3.1-3. The power levels are reduced by 2 watts from the initial day and the electrical heat contingency is omitted. The total daily energy requirement is 219 watt-hours. This power profile requires only 15 square feet of solar array and a battery capacity of 115 watt-hours.

The electrical power and energy requirements for each successive diurnal cycle utilizing the direct mode of communication are graphically illustrated by the power profile of Figure 4.3.2.3.1-4. The low level power is reduced by 5 watts from the relay communication operation due to the elimination of continuous wind measurements and the difference in the telemetry power requirement for relay and direct communications. The peak power is increased due to increased transmitter power for direct communications. The total daily energy requirement is 267 watt-hours. This power profile requires 20 square feet of solar array and a battery capacity of 154 watt-hours.

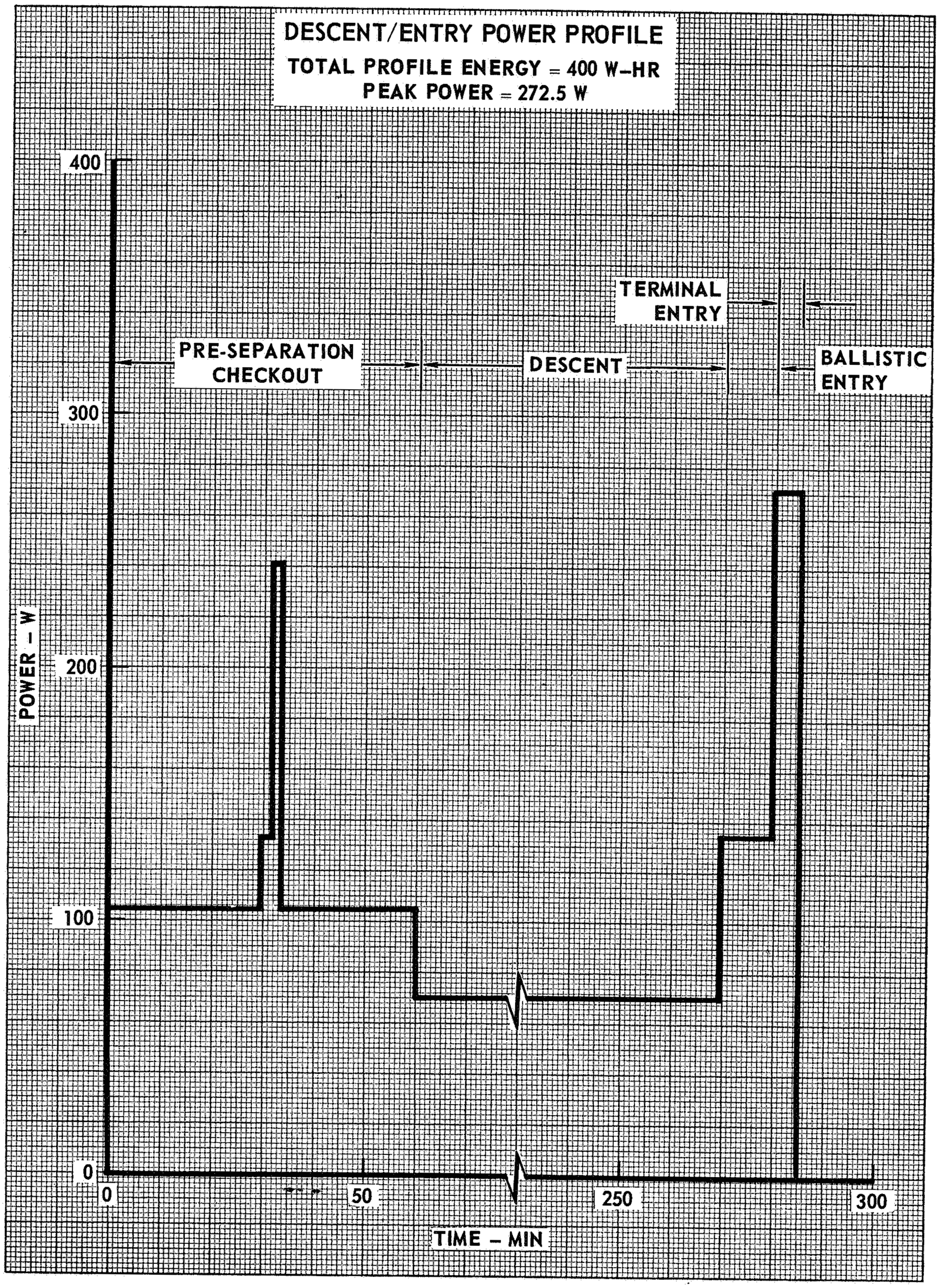


FIGURE 4.3.2.3.1–1

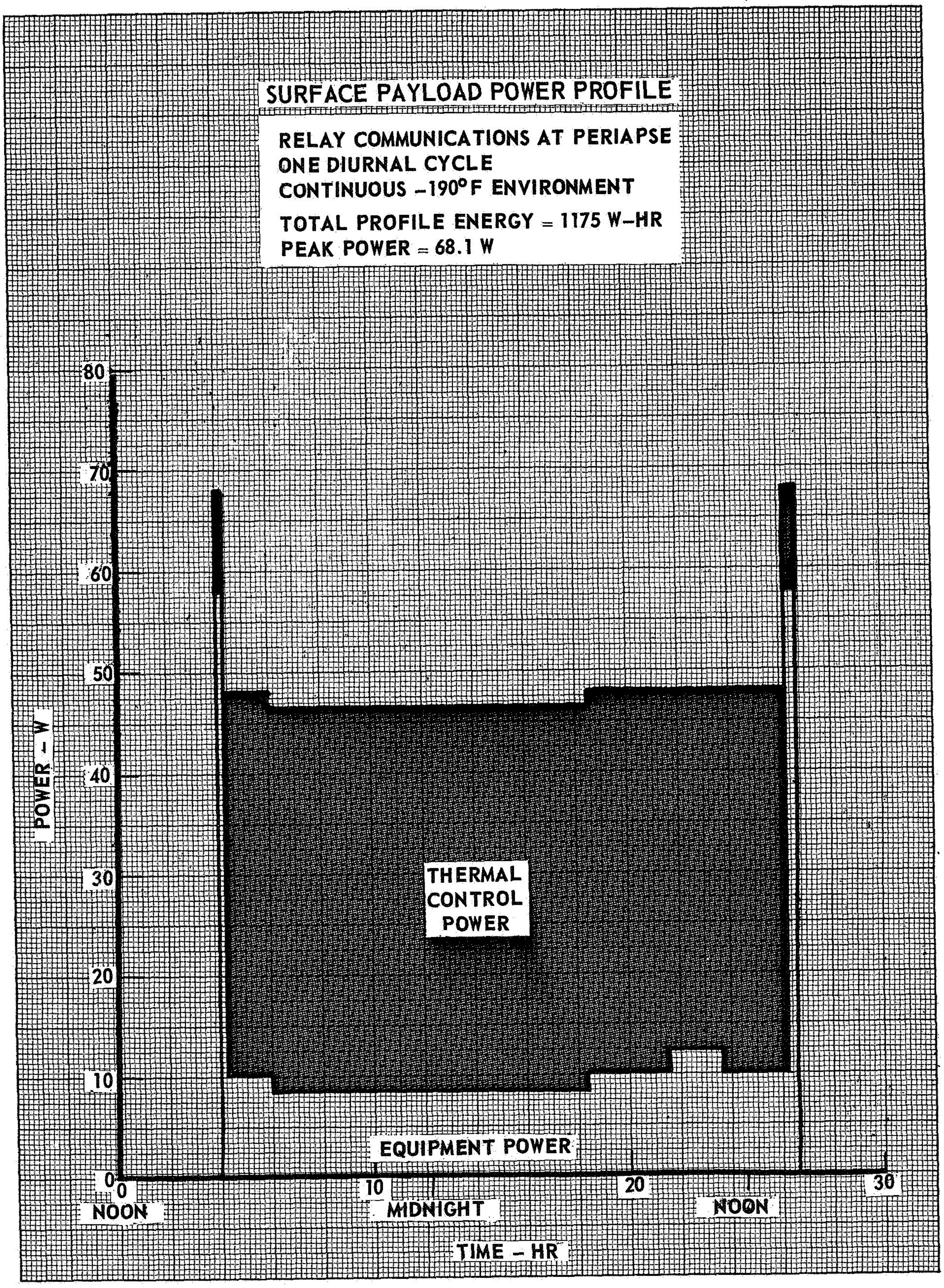


FIGURE 4.3.2.3.1-2

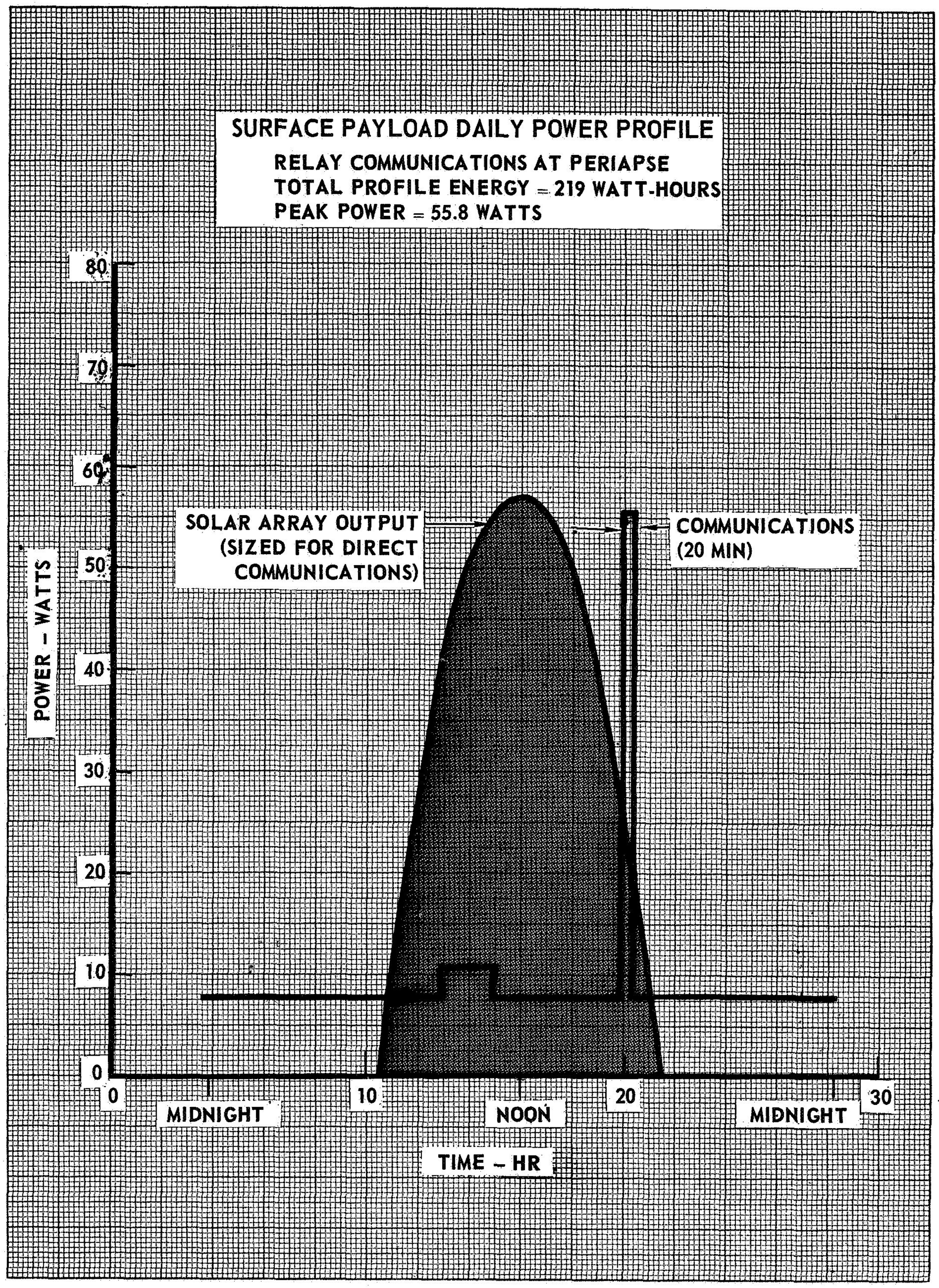


FIGURE 4.3.2.3.1-3

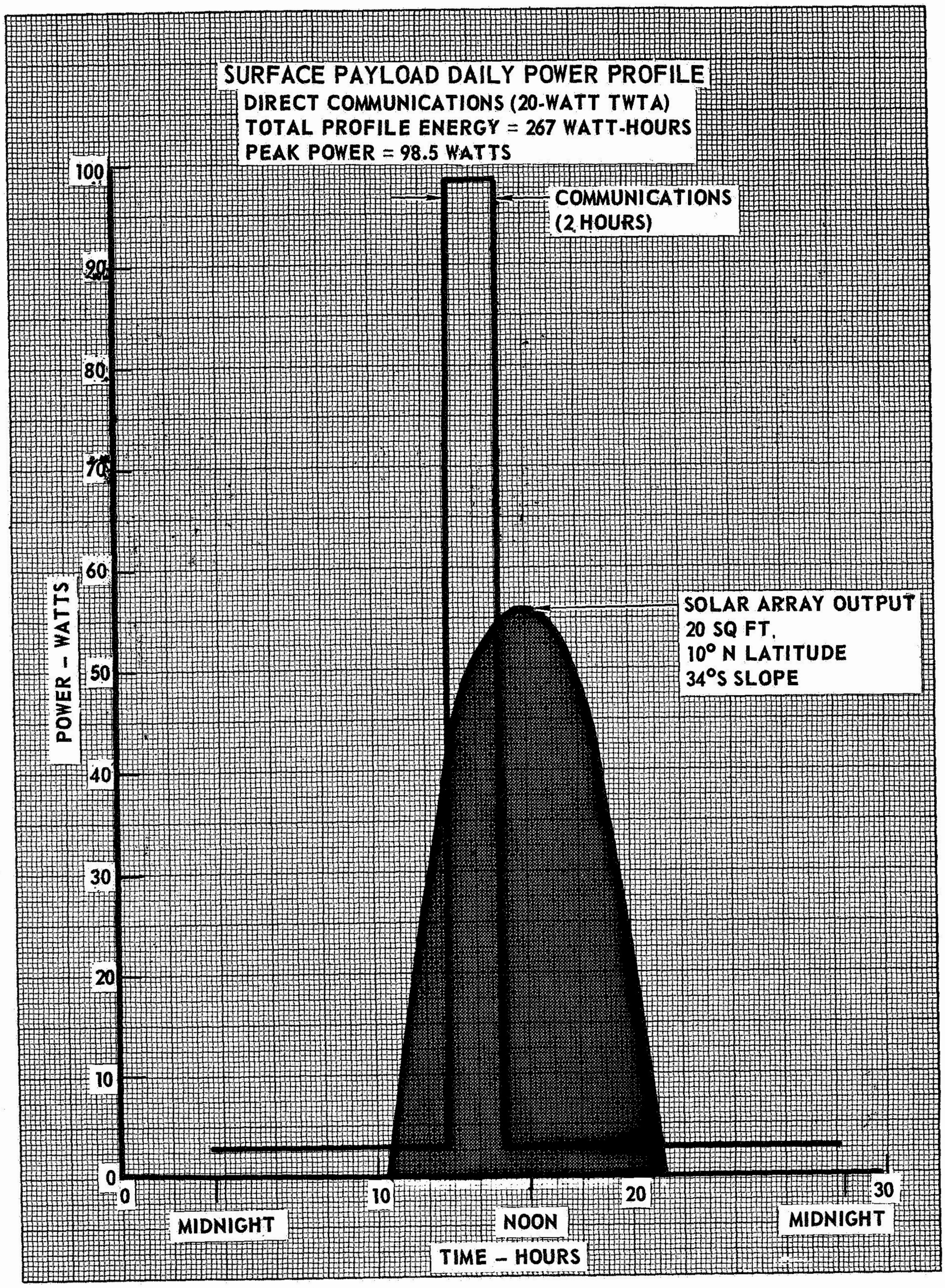


FIGURE 4.3.2.3.1–4

4.3.2.3.2 System description: The electrical power system consists of two sealed, manually activated, silver-zinc primary batteries, three automatically activated, silver-zinc batteries, two battery chargers, one DC-to-DC converter-regulator, a power switching and logic unit, and a solar cell array. A block diagram of the power system is shown in Figure 4.3.2.3.2-1. The power system weight and location are as follows:

ITEM	LOCATION	WEIGHT (LBS)
Battery Charger (2)	Canister & Surface Payload	6.0
Auto Activated Battery (3)	Lander	15.0
Manually Activated Battery	Lander	22.0
Switching & Logic	Lander & Surface Payload	13.0
Manually Activated Battery	Surface Payload	37.1
DC/DC Converter	Surface Payload	4.0
Solar Panel	Surface Payload	30.0
	Total	127.1

The DC-to-DC converter accepts unregulated power from both the orbiter solar array and the surface payload solar array, and provides regulated power to the power transfer switch in the power switching and logic unit. The power transfer switch contains the power sensing and transfer equipment associated with transfer of the distribution bus from solar array power to battery power when solar array power is not available or when commanded by the sequencer. The distribution bus and associated switches provide power to the using subsystems on commands from the sequencer.

The two, sealed, manually-activated, silver-zinc batteries are the main energy storage for the lander and surface payload. The descent/entry battery is designed for a high discharge rate to provide the power for descent and entry while maintaining adequate voltage regulation during the peak power demand of entry. In addition, this battery provides the power for the telemetry, instrumentation, and thermal control subsystems when orbiter power is not available during interplanetary cruise.

LANDER POWER SYSTEM BLOCK DIAGRAM

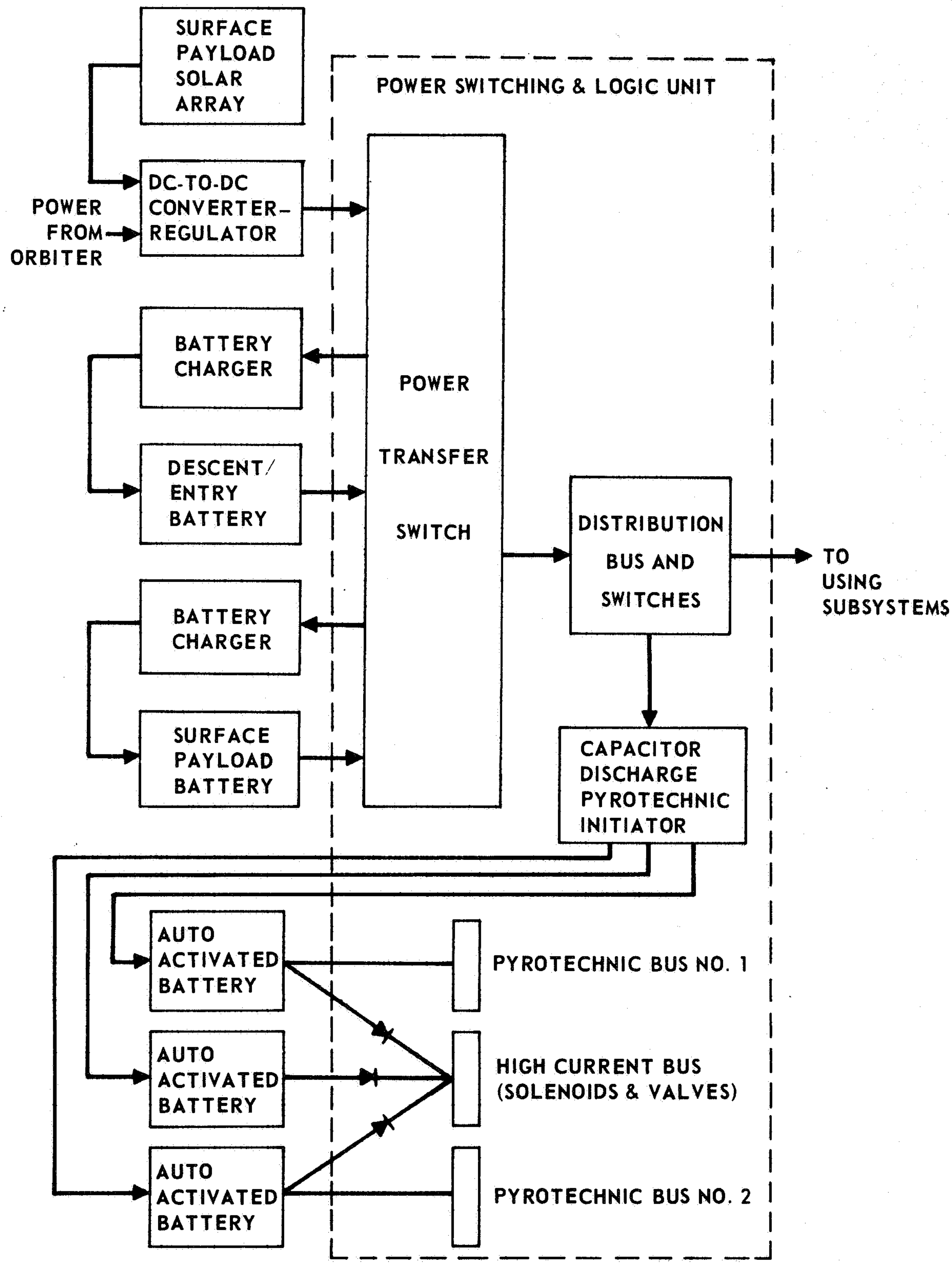


FIGURE 4.3.2.3.2-1

The surface payload battery is designed to provide the power for surface operation at a relatively low rate of discharge. This battery is sized to provide energy for one diurnal cycle of -190°F environment without any contribution from the surface payload solar array. The resulting capacity is over eight times the capacity required for subsequent days of cyclic environment and provides considerably more than adequate depth of discharge for the 90-day cycle life.

The surface payload battery must survive the Martian night environment, and thus is located in the insulated isotope heated compartment of the surface payload to maintain its temperature above 50°F. The battery charger, the DC-to-DC converter, that portion of the power switching and logic unit associated with surface payload operations, and equipment of other subsystems producing heat during surface operations are also located in this compartment.

The descent/entry battery is maintained in a charged condition during cruise and is recharged after orbiter maneuvers by its battery charger operating from orbiter power. The surface payload battery is also maintained in a charged condition during cruise by its battery charger operating from orbiter power, and is recharged during surface operations by this charger operating from the surface payload solar array power.

Each battery charger has two modes of operation. In the first mode it charges the battery at a constant potential of 1.98 volts/cell but with current limiting at a 20-hour rate. Charging current is sensed, and when this current reduces to the 100-hour rate, the battery is put on float charge at 1.87 volts $\pm$ .01 volts per cell. Each time power is removed from the battery charger and then reapplied, the charger reverts to the first operating mode.

Three automatically activated silver zinc batteries are also contained within the electrical power system. These batteries are designed to be heat sterilized in a dry charged condition. Each battery is activated using a gas generator containing dual pyrotechnic initiators. Activated stand life is at least 12 hours. The batteries are a high rate design and are used during descent and entry to supply the high pulse current requirements for the attitude control and propulsion subsystems, pyrotechnic initiators, and deployment actuators.

It is assumed that power, for operation of the lander support equipment mounted in the orbiter, will be provided directly from the orbiter power system. This equipment requires a maximum of 13.1 watts.

4.3.2.3.3 Operational description: During the pre-launch phase and until about 5 minutes before lift-off, the launch complex operational support equipment provides power to the electrical power system. During the launch phase, power is supplied from the descent/entry battery for operation of the instrumentation and telemetry subsystems. Transfer to orbiter power occurs upon completion of the launch phase and after deployment of the orbiter solar array. Subsequently, and during interplanetary cruise, orbiter power is regulated by the DC-to-DC converter and distributed to the using subsystems and battery chargers by the power switching and logic unit.

During periods of high power usage or midcourse maneuvers, orbiter power may not be available to the lander and surface payload. At this time, on command from the orbiter, and failing that upon detection of loss of orbiter power, the distribution bus is transferred to descent/entry battery power by the power switching and logic unit to maintain operation of the instrumentation and telemetry subsystems. When orbiter power becomes available again, the distribution bus is again supplied by regulated orbiter power via the DC-to-DC converter, and the descent/entry battery is again placed on float charge. The maximum amount of energy removed from the descent/entry battery during each of these periods is not expected to exceed 10 percent of its capacity.

Approximately one hour prior to separation, on command from the sequencer, the distribution bus is transferred from orbiter power to the descent/entry battery. From this time (pre-separation checkout), until landing, power is provided by the descent/entry battery.

The three automatically activated batteries are initiated, on command from the sequencer, during pre-separation checkout. Two of these batteries provide power to separate, redundant buses for pyrotechnic initiation. All three batteries are connected to a third bus for operation of high pulse current equipment.

After landing, on command from the sequencer, the distribution bus is transferred from the descent/entry battery to the surface payload solar array and the surface payload battery. The DC-to-DC converter, operating in a maximum power tracking mode, regulates the surface payload solar array output to provide power to the distribution bus for the using subsystems and, if available, battery charging power. During communication periods, more power is required than produced by the solar array. Consequently, battery charging is discontinued, and the surface payload battery is paralleled with the regulated solar array output. At night, or when solar array output is curtailed, the battery supplies all the power, and is recharged the next day.

Surface operations will continue until the surface payload battery capacity is exhausted due to extended cycling of the battery or until battery capacity is exhausted due to reduced solar array output (from any degradation source) such that the battery is not fully recharged each day. Battery cycle life is adequate for over 90 days of operation with the installed capacity for one diurnal cycle without solar array output. Solar array output is adequate for a year of operation except for degradation from sand, dust, clouds, or meteorites.

4.3.2.3.4 Performance: The performance characteristics of the power sources are tabulated in Table 4.3.2.3.4-1. Additional power system equipment is two battery changes (3 pounds each), a convertor/regulator (4 pounds), and the power switching and logic unit (13 pounds). The power switching and logic unit will be subdivided into two sections: one for descent and entry equipment; one for surface payload equipment.

The performance characteristics of the manually activated batteries are predicated on the projected energy density of Figure 3.2.3-7 and the projected cycle life of Figure 3.2.3-9. Therefore the performance is necessarily conservative and assumes continuing development of heat sterilizable silver-zinc batteries.

The possible daily solar array output performance is shown in Figure 3.2.3-32. With the 24 May 1974 landing date, a 34° adverse slope, and a 90-day

TABLE 4.3.2.3.4–1

POWER SOURCE PERFORMANCE CHARACTERISTICS

CONCEPT III

DESCENT/ENTRY BATTERY

TYPE = PRIMARY; SEALED, MANUALLY ACTIVATED, SILVER-ZINC
CAPACITY = 425 WATT-HOURS
ENERGY DENSITY = 19 WATT-HOURS/POUND
WEIGHT = 22 POUNDS
WET STAND LIFE = 12 MONTHS
CYCLE LIFE = 4 CHARGE/DISCHARGE CYCLES AT 100% OF CAPACITY
REGULATION = 29.5 ± 4.5 VOLTS
TEMPERATURE LIMITS:
WET STORAGE = 0°F TO 60°F
NORMAL USE = 50°F TO 120°F

SURFACE PAYLOAD BATTERY

TYPE = SECONDARY; SEALED, MANUALLY ACTIVATED, SILVER-ZINC
CAPACITY = 1245 WATT-HOURS
ENERGY DENSITY = 34 WATT-HOURS/POUND
WEIGHT = 37 POUNDS
WET STAND LIFE = 15 MONTHS
CYCLE LIFE = 90 CHARGE/DISCHARGE CYCLES AT 50% OF CAPACITY
REGULATION = 28 ± 4 VOLTS
TEMPERATURE LIMITS:
WET STORAGE = 0°F TO 60°F
NORMAL USE = 50°F TO 120°F

HIGH CURRENT BATTERY

TYPE = PRIMARY; AUTOMATICALLY-ACTIVATED, SILVER-ZINC
CAPACITY = 20 WATT-HOURS
ENERGY DENSITY = 4 WATT-HOURS/POUND
WEIGHT = 5 POUNDS
WET STAND LIFE = 12 HOURS
DRY STORAGE LIFE = 24 MONTHS
REGULATION = 27 ± 7 VOLTS
PEAK LOAD CURRENT = 25 AMPERES
TEMPERATURE LIMITS:
UNACTIVATED = –65°F TO 160°F
ACTIVATED = 20°F TO 120°F

SOLAR CELL ARRAY

TYPE = N/P SILICON CELLS
CONFIGURATION = FLAT, FIXED ARRAY
SIZE = 20 SQUARE FEET
WEIGHT = 30 POUNDS
OUTPUT (10°N LATITUDE):
34° ADVERSE SLOPE = 410 WATT-HOURS DAY
34° FAVORABLE SLOPE = 733 WATT-HOURS DAY

mission, the design point is 20.5 watt-hours/day/square foot and occurs in August for a 34°S slope. However, for lessor slope angles, for any other slope direction, for the landing date, and for mission extension beyond 90 days, the output will be considerably better than the design point.

4.3.2.3.5 Development status: Except for the batteries, the electrical power system components are not long lead time development items. However, battery development is critical for a 1973 launch date. Maximum development, effort is required in order to obtain adequate engineering data of wet life, cycle life, and loss rates on heat sterilizable batteries. Currently, little data exists on wet life and cycle life of heat sterilizable systems. Data is urgently needed on the cycle life capability after the extended wet stand period of interplanetary cruise. This data is required in order to size and plan battery capacity for proper depth of discharge to provide the cycle life required for a specific mission. Since the interplanetary cruise period is about 220 days, and a 90-day surface mission capability is desirable, a program to provide this data will require at least a year or more of testing before significant data is obtained. Therefore, immediate initiation of such a program is imperative to provide adequate data for design of a vehicle for the 1973 launch date.

An alternate solution to wet heat sterilization of sealed, Ag-Zn batteries is sterile activation of a dry, heat-sterilized battery and electrolyte and subsequent sterile insertion or assembly into the flight capsule. This technique has been proven feasible with no failure of the sterilization procedure. Furthermore, long wet life, silver-zinc batteries of current, non-sterile technology were tested, indicating that little development may be required to extend current technology to a sterile battery by this dry heat sterilization technique.

4.3.2.4 Guidance and control. - The guidance and control system consists of attitude/deorbit velocity control electronics (inertial measurement unit, and central computer and sequencer), landing radar, and radar altimeter.

4.3.2.4.1 Attitude/deorbit velocity control electronics: This system consists of an Inertial Measurement Unit (IMU) and Central Computer and Sequencer (CC&S) with their power supplies. The IMU senses inertial body rates about all three body axes and acceleration along the roll axis. The CC&S accepts and processes the IMU signals and landing radar range and velocity measurements and generates commands for the propulsion subsystems to guide and control the capsule from orbiter separation to touchdown on the Martian surface. The CC&S also performs all capsule and lander sequencing. The design selection of the control sensors, computer, and flight sequencing has been derived from parametric studies presented in Sections 3.2.4 and 3.2.5. The weight savings resulting from this design approach is discussed in detail in Section 3.2.5. The guidance and control equipment total weight is 37 lb.

Control Requirements and Design Goals - Table 4.3.2.4-1 summarizes the control system operations during each mission phase. Table 4.3.2.4-2 shows the control accuracy design goals and calculated system performance. The system performance in all cases meets or exceeds the requirements and design goals.

The 300 meter separation required between the orbiter and the capsule before deorbit motor ignition is accomplished within 20 minutes, to limit the system attitude drift before this critical event. Systems checkout capability after enclosure in the sterilization canister is provided both on the pad and during flight.

System Description - A functional block diagram of the attitude/deorbit velocity control electronics is shown in Figure 4.3.2.4-1, and a summary of characteristics is given in Table 4.3.2.4-3. The strapped-down IMU with a digital computation capability provides a minimum weight and risk system and also the greatest mission accuracy and flexibility.

The IMU contains three rate integrating gyros, one integrating accelerometer, and a power supply. This unit is mounted on the lander, near the capsule center of mass, such that the accelerometer senses decelerations along the

TABLE 4.3.2.4-1

ATTITUDE DEORBIT VELOCITY CONTROL SYSTEM OPERATIONAL SEQUENCE

MISSION PHASE	SYSTEM OPERATIONAL DESCRIPTION
PRIOR TO SEPARATION	• WARM-UP AND ACTIVATION. • OPERATIONAL VERIFICATION • ALIGNMENT TRANSFER AND SEPARATION MODE SETUP.
SEPARATION	• CAPTURE PRE-SEPARATION ALIGNMENT. • 3-AXIS ATTITUDE HOLD TO ± 0.25 DEG
COAST-TO-DEORBIT	• MANEUVER CAPSULE INTO DEORBIT THRUST ATTITUDE; REFERENCE FRAME IS ROTATED, ONE AXIS AT A TIME AT 1.0 DEG/SEC, WITH VEHICLE FOLLOWING. • RETURN TO THREE AXIS HOLD MODE, ± 0.25 DEG.
DEORBIT	• CONTINUE THREE AXIS HOLD MODE, ± 0.25 DEG. • TERMINATE THRUST USING INTEGRATED AXIAL ACCELEROMETER SIGNAL.
COAST-TO-ENTRY	• MANEUVER TO ZERO ENTRY ANGLE OF ATTACK ATTITUDE. SIMILAR TO DE-ORBIT THRUST MANEUVER. • SWITCH TO 3-AXIS ATTITUDE HOLD, ± 3 DEG.
ENTRY	• SENSE ATMOSPHERIC ENTRY (0.05g) • SWITCH TO PITCH AND YAW RATE LIMITING (± 3 DEG/SEC). • CONTINUE ROLL ATTITUDE HOLD (± 3.0 DEG) • INITIATE AERODYNAMIC VELOCITY VECTOR DIRECTION CALCULATION.
PARACHUTE DESCENT	• UNCONTROLLED FLIGHT. • CONTINUE VELOCITY VECTOR DIRECTION CALCULATIONS.
TERMINAL DESCENT	• PARACHUTE SEPARATION AND TERMINAL ENGINE IGNITION. • 3-AXIS ATTITUDE HOLD, ROLL AXIS ALIGNED TO COMPUTED AERODYNAMIC VELOCITY VECTOR FOR RADAR ACQUISITION. • CONSTANT AXIAL ACCELERATION COMMAND. • AFTER RADAR ACQUISITION, ROTATE REFERENCE FRAME WITH VEHICLE FOLLOWING TO NULL LATERAL VELOCITY COMPONENTS. • INTERCEPT PRE-PROGRAMMED DESCENT PROFILE AND GENERATE VELOCITY COMMANDS AS FUNCTION OF SLANT RANGE. • AT PRE-PROGRAMMED SLANT RANGE SWITCH BACK TO 3-AXIS ATTITUDE HOLD, AND COMMAND CONSTANT VELOCITY DESCENT TO ENGINE CUT-OFF.

TABLE 4.3.2.4–2

SUMMARY OF SYSTEM DESIGN GOALS & PERFORMANCE

CONTROL PARAMETER	DESIGN GOAL (3σ)	SYSTEM PERFORMANCE (3σ)
• DE-ORBIT THRUST DIRECTION POINTING ERROR	± 1.0°	± 0.81°
• DEORBIT VELOCITY (ΔV) INCREMENT UNCERTAINTY		
– PARALLEL TO SPECIFIED COMPONENT	± 0.75%	± 0.5%
– NORMAL TO SPECIFIED COMPONENT	± 1.5%	± 1.5%
• ENTRY ANGLE-OF-ATTACK UNCERTAINTY (10 HR MAX. DESCENT TIME)	± 15.0°	± 13.6°
• ROLL REFERENCE UNCERTAINTY AT ENTRY (10 HR MAX DESCENT TIME)	± 15.0°	± 13.6°
• VELOCITY VECTOR DIRECTION UNCERTAINTY AT TERMINAL PROPULSION IGNITION	± 20°	± 17°

ATTITUDE/DEORBIT VELOCITY CONTROL ELECTRONICS
BLOCK DIAGRAM

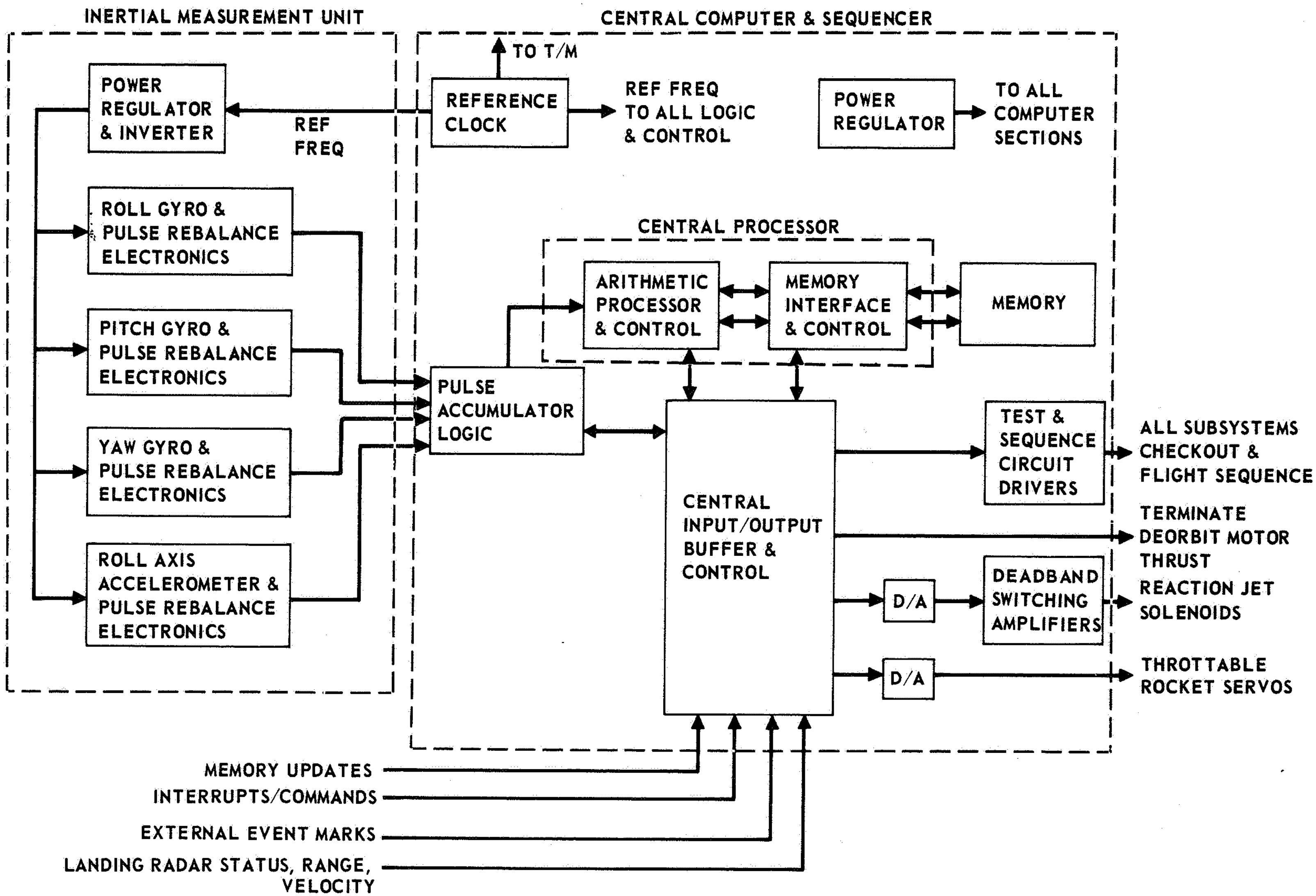


FIGURE 4.3.2.4-1

TABLE 4.3.2.4–3

SUMMARY OF ATTITUDE DEORBIT VELOCITY CONTROL ELECTRONICS CHARACTERISTICS

INERTIAL MEASUREMENT UNIT	CENTRAL COMPUTER & SEQUENCER
GYROS • HONEYWELL GG334S • SINGLE DEGREE-OF-FREEDOM, RATE INTEGRATING • PULSE-ON-DEMAND REBALANCE ELECTRONICS • · 1.0 DEG/HR (3σ) g-INSENSITIVE DRIFT • · 1.5 DEG/HR g (3σ) MASS UNBALANCE DRIFTS • · 60 DEG/SEC MAX TORQUEING RATE (ESTIMATED MAX VEHICLE RATE 30 DEG/SEC) ACCELEROMETER • HONEYWELL GG177 • PULSE-ON-DEMAND REBALANCE ELECTRONICS • $\pm 150 \times 10^{-6}$ g (3σ) BIAS • ± 0.015% (3σ) SCALE FACTOR STABILITY • 0 TO 30 g's RANGE SIZE • 375 CU IN. WEIGHT • 16.9 LB POWER • 40 W • ESTIMATED 15 W AVERAGE THERMAL CONTROL POWER	TYPE • MAGNETIC CORE • SERIAL ARITHMETIC CAPACITY • 4096 WORD MEMORY • 20 BIT DATA WORD • 512 WORDS UPDATABLE SPEED • 14 μSEC ADD TIME • 154 μSEC MULTIPLY TIME INSTRUCTIONS • 25 CAPABILITY • PROVIDES ALL TEST AND FLIGHT SEQUENCING • INTERRUPT CAPABILITY • ANALOG OUTPUT TO PROPULSIVE UNITS SOLENOIDS/SERVOS SIZE • 555 CU IN. WEIGHT • 20.1 LB POWER • 34 W • 15.9 W DURING LONG DESCENT PHASE BY SPECIAL LOGIC IDLE MODE

roll axis and the gyros sense rotations about each orthogonal body axis. A single accelerometer, mounted on the roll axis, is adequate to meet the deorbit velocity accuracy requirement. Each IMU sensor uses the ternary (pulse-on-demand) rebalance principle. This technique provides lower biases while also allowing higher dynamic ranges. Each pulse from a gyro or the accelerometer represents, respectively, a precise incremental attitude or velocity change.

The CC&S is a general purpose digital computer with a central processor, core memory, control circuitry, and gating and interface circuitry. A power conditioner, with special guard against memory loss, is self-contained. This package is mounted on the lander, next to the IMU. A memory sizing study showed a need for 1850 permanent storage data words and program instructions and 426 variable data words. A 4096 word capacity (of which 512 are updatable) is provided for mission growth. This capacity also provides for all pre-separation checkout testing and landing flight sequencing. The data word of 20 bits is based on simulation studies which show 19 bits are adequate to limit round off errors in the linearized Euler dynamic equations used for attitude stabilization. The other bit is provided for arithmetic sign. The operational speed and serial arithmetic was selected after establishing the maximum iteration speeds for the attitude control and flight sequencing subroutines.

Operational Description - The operational sequence is shown in Figure 4.3.2.4-1. Approximately 24 hours prior to capsule/orbiter separation, the CC&S is activated. All systems and built-in test equipment are exercised according to the CC&S stored memory sequence. The first test function verifies the CC&S memory content, and checks specific test algorithms. The IMU sensors are monitored while operating to detect unacceptable gyro drifts, accelerometer bias, and scale factor instability. The system is also checked end-to-end (i.e., from the gyro response to the propulsion element driver signal).

One hour prior to separation, the IMU is activated under the control of the CC&S incorporated landing flight sequencing program. The separation attitude reference is initialized in the CC&S. After separation, the control system maintains a three axis attitude hold ($\pm$0.25 deg) using body attitude error summed with body rate in commanding the Reaction Control System (RCS) operation. The computer calculated rate stabilizes the attitude control by providing active damping.

During the coast-to-deorbit phase, the capsule is maneuvered to the deorbit thrust attitude. This control mode is identical to the preceding, except that the reference frame is rotated, one axis at a time, at a constant 1.0 deg/sec rate. The vehicle axes follow the rotated reference frame. After maneuver termination, the system returns to the $\pm$0.25 degree, three axis attitude hold mode for deorbit thrusting. Thrusting is terminated using the integrated axial accelerometer output.

After deorbit thrusting, the capsule is maneuvered into a zero entry angle of attack attitude. This maneuver mode is identical to that used for the deorbit thrust maneuver. During the period between maneuver completion and entry, computer calculated attitude stabilization is employed. A three axis attitude hold deadband of $\pm$3 degrees results in minimal limit cycle fuel consumption and control accuracy degradation. The computer accumulates IMU attitude change pulses, calculates inertial body rates, and generates RCS jet commands from error signals derived from linearized Euler angle dynamic equations. These attitude calculations eliminate any "coning" errors resulting from limit cycle motions. The low iteration speeds of the subroutine during this phase allow the computer logic to be cycled on/off. This logic idle technique reduces the computer power consumption by 53%.

At atmospheric entry (0.05g) attitude control is switched to pitch and yaw rate damping ($\pm$3 deg/sec rate limiting). The roll attitude hold ($\pm$3.0 deg) is continued. Rate damping in the pitch and yaw axes assists in maintaining aerodynamic trim, and provides a stable environment for the entry experiments. The preferred roll mode allows the use of a fan beam radar altimeter with significant accuracy, range, and weight improvements relative to an altimeter system with a hemispherical radiation pattern.

Calculation of the aerodynamic velosity vector direction begins at atmospheric entry. During entry at peak dyanmic pressure, the capsule roll axis is closely aligned to the velocity vector. At this time a computer routine is initiated, and the computer begins the calculation of the direction cosines of the body axes. The routine effectively filters the computed pitch and yaw Euler angles, and uses them to rotate the reference frame. In this manner the computed roll axis reference follows the gravity turn without following the higher frequency body oscillations caused by winds and other disturbances. The angles derived from these computed direction cosines become the initial pitch and yaw Euler angles used at the start of terminal descent. This technique provides a favorable attitude for landing radar acquisition.

A radar altimeter mark initiates the parachute deployment/aeroshell separation sequence. During parachute descent no reactive control is employed. The calculation of the aerodynamic velocity vector direction is continued. Terminal descent engine ignition and parachute release is initiated by a signal from the radar altimeter.

At the beginning of terminal descent an inertial attitude hold mode is established, using the computed pitch and yaw Euler angles from the velocity vector direction routine. In the attitude hold mode, attitude error and body rate are summed to form proportional throttle commands. A constant axial acceleration command with an acceleration feedback loop closed through the IMU is used.

After landing radar acquisition, the attitude control is switched from inertial attitude hold to radar steering in the pitch and yaw body axes to null the radar measured lateral velocity components. Upon intercepting the pre-programmed descent profile, the command of the axial component of the terminal propulsion thrusters is generated according to the pre-programmed range/velocity function. At a pre-set slant range (e.g., 50 ft), attitude control is switched back to inertial hold, and a constant velocity (e.g., 5 ft/sec) is commanded until engine cutoff at 10 ft altitude.

Performance - System performance is presented in Table 4.3.2.4-2 along with design goals. In addition to all dynamic error sources, the attitude accuracy error analyses have also included all irreducible initial system errors. These include: (1) IMU to capsule mounting alignment uncertainty,

(2) capsule to orbiter mounting alignment uncertainty, and (3) orbiter deadband limit cycle motions at capsule separation. The calculated capsule control performance meets or excels all requirements and design goals.

Development Status - The digital IMU will incorporate sterilizable sensors. The selected gyro has been developed under a special sterilization design/development contract. The accelerometer has been built, tested, and successfully operated in temperature environments which are as stringent as those imposed by sterilization. Neither type of sensor poses ETO test cycle problems, since they are both hermetically sealed. The concept of body mounted, pulse rebalanced sensors has been successfully used for the PRIME system mission, which has environmental extremes exceeding those expected for this capsule. The LM abort sensor assembly (IMU) containing pulsed rebalanced gyros and accelerometers has successfully completed qualification testing.

The computer can be mechanized from existing components and technology. Similar magnetic core systems have been test proven to withstand the sterilization environment. The magnetic core memory technology has been space qualified in numerous previous applications.

4.3.2.4.2 Landing radar system: The landing radar system employs three CW (continuous wave) velocity sensing beams and a single linear FM/CW (frequency modulation/continuous wave) range sensing beam. This design is based on a minimum modification to the LM (Lunar Module) landing radar. Landing radar derived range and velocity information is provided from parachute separation altitude to engine cut-off. Parametric study results of the landing radar system are discussed in Section 3.2.4.2.

Requirements and Constraints - The primary requirements of the landing radar are as follows:

a) To provide continuous roll axis slant range and 3-axis velocity vector measurement from parachute separation to engine cut-off (6500 to 10 ft.).

b) To provide terminal phase measurement errors (3σ) of less than $\pm 1.4\%$ $+5$ ft. for slant range and $\pm 1.5\%$ ± 1.5 fps for the resolved velocity components.

c) To provide capability for checkout of the radar after canister enclosure and during flight.

System Description - A block diagram of the landing radar is shown in Figure 4.3.2.4-2, and a summary of characteristics is given in Table 4.3.2.4-4. This landing radar is based on a minimum modification to the existing LM landing radar. This radar employs three CW velocity sensing beams and a single sawtooth FM/CW range beam. The velocity beams are positioned at half cone angles of 25 degrees from the roll axis and at clock angles of 0, 110, and 180 degrees. The range beam is positioned along the lander roll axis between the 0 and 180 degree velocity beams. Two solid state transmitters are used, one each for the velocity and ranging functions. A single integrated planar array antenna contains interlaced velocity and range transmit arrays together with the four receive arrays.

Each of the four receive channels employs quadrature homodyne detection to facilitate sense determination and descrimination against double sideband spurious signals. After pre-amplification, the beat frequencies are supplied to frequency trackers. The tracker output frequencies are proportional to the doppler or range frequencies (plus a constant offset) on each beam. The velocity tracker signals are resolved into orthogonal components, converted to serial binary form, and supplied to the guidance and control system. Reliable lock and operation signals are generated to indicate individual tracker, as well as range and velocity sensing, operational status. A CRO (Conditional Reliable Operation, see Section 3.2.4.2) mode is available if near zero doppler situations are encountered. The electronics assembly provides for self-testing by generation and injection of test signals into the preamplifiers.

Physical Description - The 38.6 lb landing radar system is divided into the antenna assembly and electronics assembly. The antenna assembly consists of a single interlaced transmitter array, four receiver arrays, velocity and range transmitters, mixer crystal assemblies, and audio frequency amplifiers. The 20 x 24.6 x 6.5 inch antenna assembly is located on the underside near the center of the lander footpad, resulting in a clear field of view. The

BLOCK DIAGRAM OF LANDING RADAR

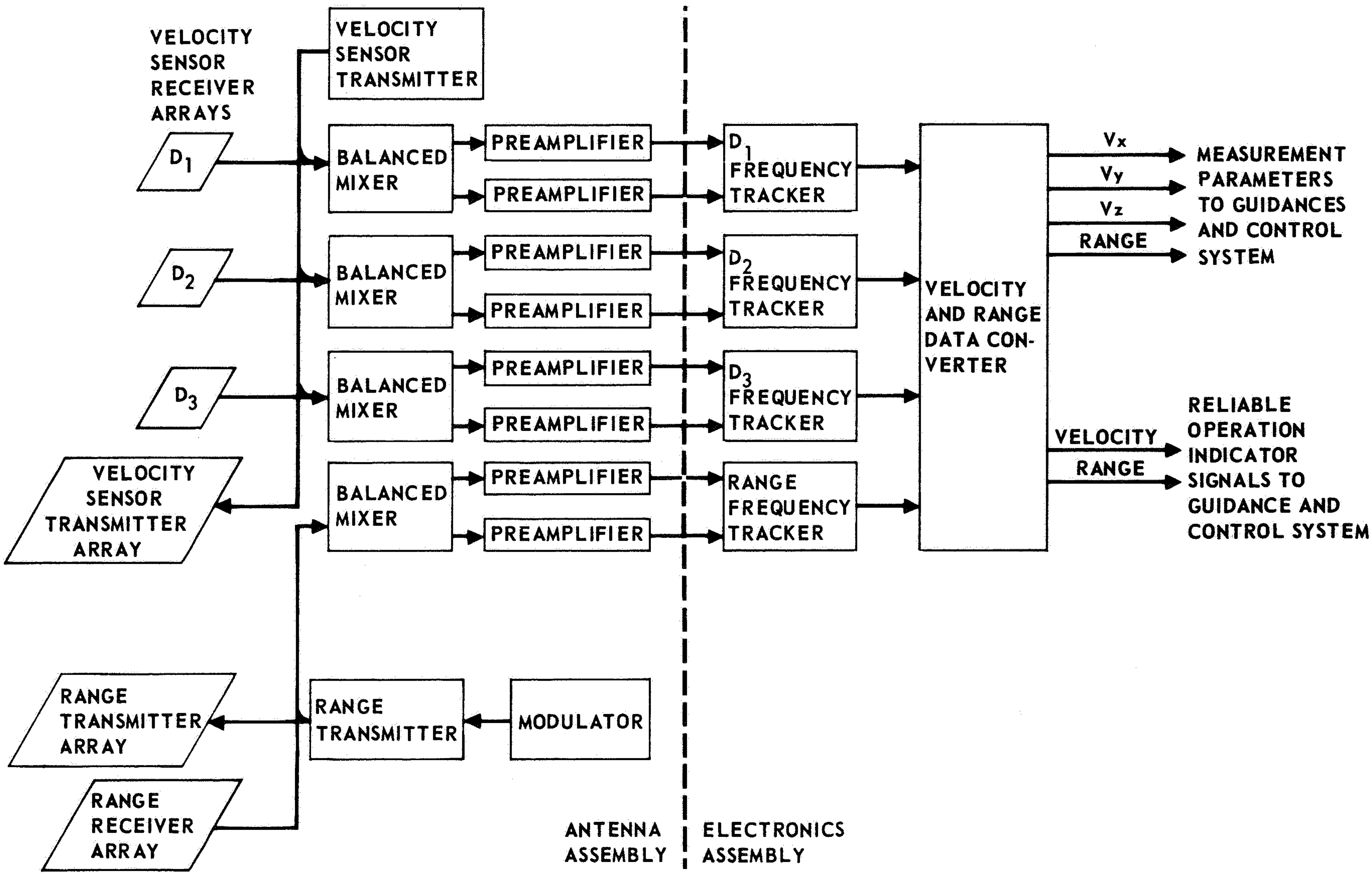


FIGURE 4.3.2.4-2

TABLE 4.3.2.4–4

SUMMARY OF LANDING RADAR CHARACTERISTICS

FREQUENCY	
RANGE BEAM OPERATING FREQUENCY	9.58 GHz
VELOCITY BEAMS OPERATING FREQUENCY	10.51 GHz
ANTENNAS	
RANGE BEAM ANTENNA GAIN	26.7 dB
VELOCITY BEAMS ANTENNA GAIN	27.6 dB
RANGE ANTENNA BEAMWIDTH (2-WAY)	3.9 x 7.5 DEGREES
VELOCITY ANTENNA BEAMWIDTHS (2-WAY)	3.7 x 7.3 DEGREES
RANGE BEAM POSITION	ALONG ROLL AXIS
VELOCITY BEAM POSITIONS	25 DEGREES FROM ROLL AXIS AT CLOCK ANGLES OF 0°, 70°, AND 180°
TRANSMITTERS	
RANGE BEAM RADIATED POWER	39 MW
VELOCITY BEAM RADIATED POWER	33 MW PER BEAM
RANGE BEAM MODULATION	LINEAR SAWTOOTH FM
RANGE BEAM FREQUENCY DEVIATION	8 MHz, R > 2500 FEET 40 MHz, R < 2500 FEET
VELOCITY BEAM MODULATION	NONE
RECEIVERS	
RECEIVER NOISE FIGURE	14 dB AT 10 KHz, INCREASING WITH DECREASING FREQUENCY
RANGE TRACKER FILTER BANDWIDTH	1600 Hz, R > 2500 FEET 1000 Hz, R < 2500 FEET
VELOCITY TRACKER FILTER BEAMWIDTH	1000 Hz, R > 2500 FEET 400 Hz, R < 2500 FEET
ACQUISITION TIME (4 dB SNR)	3 SECONDS (MAXIMUM)
ACCURACY	
VELOCITY ACCURACY (3σ TERMINAL PHASE)	1.5% ± 1.5 FPS
RANGE ACCURACY (3σ TERMINAL PHASE)	1.4% ± 5 FEET
SIZE, WEIGHT, POWER	
ANTENNA WEIGHT	23.3 LB
ANTENNA SIZE	24.6 x 20 x 6.5 IN.
ELECTRONICS ASSEMBLY SIZE	4.9 x 6.8 x 14.4 IN.
ELECTRONICS ASSEMBLY WEIGHT	13.3 LB
MICROWAVE ABSORBER WEIGHT	2.0 LB
PRIMARY POWER	137 WATTS

electronics assembly contains the four frequency trackers, data converters, power supply, and cabling subassembly. Size, weight, and power requirements for the two assemblies are given in Table 4.2.2.4-4. In addition, approximately two pounds of microwave absorbing material are placed on the aeroshell nose cap interior to allow radar testing after canister enclosure.

Operational Description - Prior to deorbit the landing radar is energized for self-testing. To assure adequate varactor frequency multiplier warm-up time, the radar is again energized at 100 000 ft, based on a radar altimeter mark. The radar search/track function is inhibited until after separation from the aeroshell.

After aeroshell separation and during the parachute descent phase, the radar is allowed to acquire and track the surface signals to verify proper radar operation. Landing radar measured data is not used for external control during parachute descent because the possible extreme attitudes and attitude rates prevent continuous tracking. (In the event that a radar malfunction is detected during parachute descent, the parachute may be retained to touchdown.)

At an altitude of about 6500 ft the parachute is released via a radar altimeter mark. By this time the aeroshell has reached the surface. At about the same time the descent engines are ignited, an inertial attitude hold mode is initiated, and the radar search/track mode is updated, causing the trackers to search over pre-determined frequency intervals corresponding to the conditions at 6500 ft. Within two seconds after engine ignition, the inertial attitude hold mode has established favorable attitude and attitude rate conditions for radar acquisition.

Range and velocity track is normally established within 5 seconds after engine ignition. The resolved orthogonal velocity components and the slant range measurements are supplied to the central computer and sequencer. The lander is then caused to align with the measured velocity vector. During this maneuver attitude rate limiting is employed to avoid excessive range and velocity frequency tracking rates.

The radar derived range and velocity information is used continuously during the remaining portion of the terminal descent, until engine shutdown at 10 ft altitude. During this latter portion of flight, landing radar mode switching will be performed (based on a prescribed combination of measured range and velocity) wherein tracker bandwidths are reduced and the range beam frequency deviation is increased to provide increased accuracy.

Performance - The performance characteristics of the landing radar are summarized in Table 4.3.2.4-4. These characteristics are determined by the radar beam configuration, the per beam transmitting and receiving parameters, and signal processing techniques. The terminal phase accuracies of Table 4.3.2.4-4 are based on computer studies described in Reference 4.3.2.4-1. Accuracy will be degraded during the initial phase of radar operation due to less favorable altitude, attitude, and attitude rate conditions.

Development Status - The landing radar system design is predicated on the maximum utilization of the existing LM radar system. The major modifications required to adapt it to the Mars mission are as follows:

a) Alternate Antenna System - The antenna beam configuration must be changed to locate the ranging beam at the beam group centerline. An engineering model of an antenna providing this beam configuration has been fabricated and tested.

b) Protection from Plume Damage - The heat load from the descent engines will require antenna thermal design modifications. The present slotted waveguide array technique will require RF transparent material, either bonded to the array faces or filled in individual slots, to prevent exhaust gas backflow damage.

c) Sterilization - Design modification to incorporate sterilizable components and materials is required. From Reference 4.3.2.4-1 it is estimated that 5.3% of the present LM part population is questionable with regard to sterilization.

Additional required modifications include the provision for in-flight monitoring and check-out, power supply redesign for the lander input voltage levels, tracker parameter optimization for the Martian descent conditions, and incorporation of the Conditional Reliable Operation mode.

4.3.2.4.3 Radar altimeter system: A non-coherent pulse radar altimeter system similar to system III of Section 3.2.4.3 is described herein. This system differs from the baseline system III because of the inclusion of a secondary antenna to allow operation after aeroshell separation.

Requirements and Constraints - Radar altimeter system requirements and constraints are summarized in Section 3.2.4.3. Factors of particular significance to this system are listed below.

a) It provides continuous altitude information from an altitude of 200 000 ft to lander release from the parachute for entry science data correlation.

b) It provides descrete altitude marks for event sequencing, including landing radar turn-on, parachute deployment, parachute release, and engine ignition.

c) It provides altitude accuracy consistent with the constraints of Section 3.2.4.3.

d) It provides the capability for radar checkout after canister enclosure and during flight.

System Description - The radar altimeter is a non-coherent pulse system employing leading edge tracking. A triode cavity transmitter operating at 1.0 GHz provides 5 μsec pulses with a 500 w peak power. A low noise figure superheterodyne receiver with RF amplification is used. Two separate antennas are used. The primary antenna is a single T-fed slot mounted on the aeroshell, which provides fan beam coverage during entry with the capsule held in a preferred roll position. The secondary antenna is a linearly polorized crossed slot configuration mounted on the lander and used after separation from the aeroshell. The system provides altitude information from 200 000 ft to the point of parachute release. A block diagram of the radar altimeter is shown in Figure 4.3.2.4-3 and the system characteristics are summarized in Table 4.3.2.4-5.

Referring to Figure 4.3.2.4-3, the 5 μsec, 500 w transmitter pulses are generated by the high voltage power supply and modulator pulsing the triode oscillator grid. The modulator trigger pulse train is derived from the timing

RADAR ALTIMETER – FUNCTIONAL BLOCK DIAGRAM

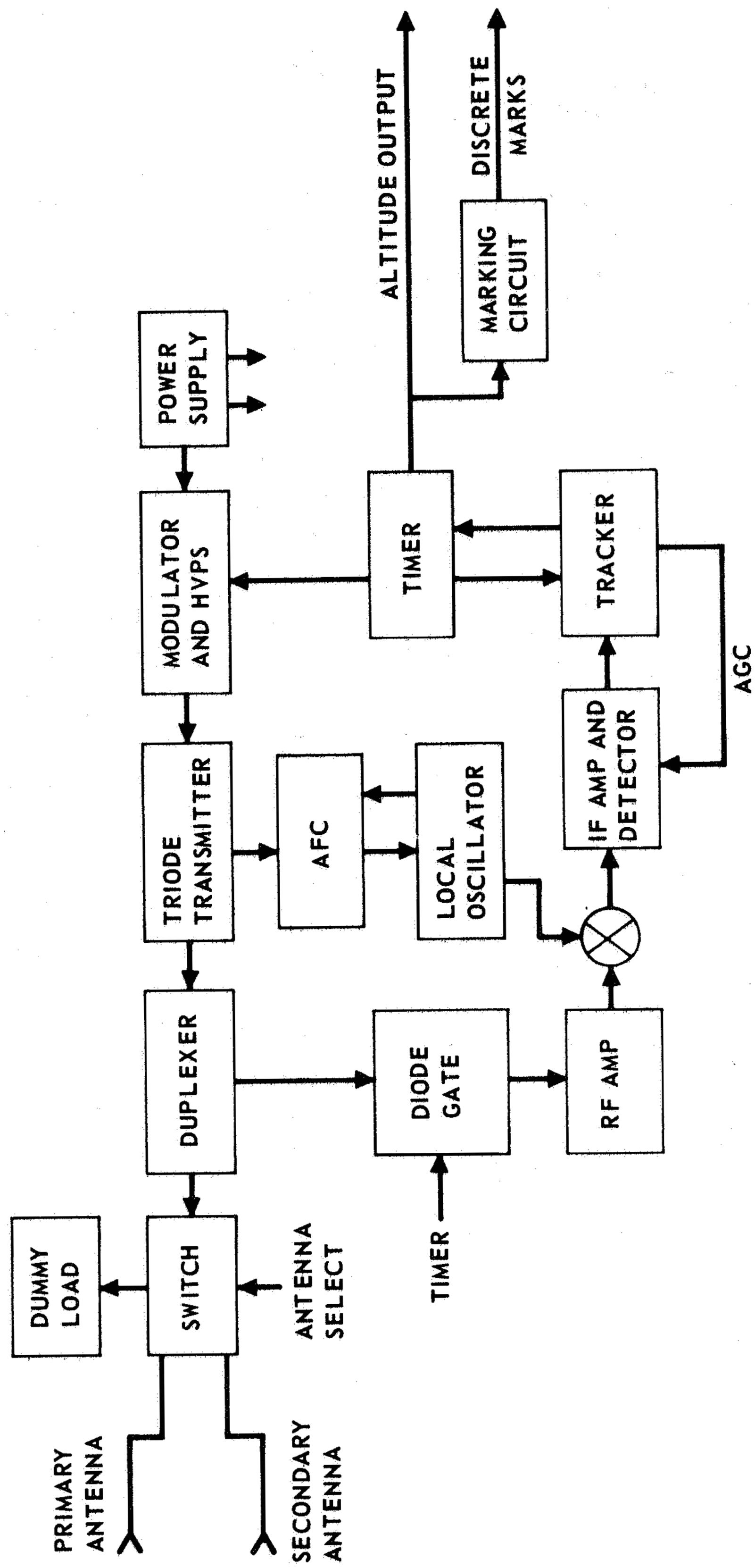


FIGURE 4.3.2.4-3

TABLE 4.3.2.4–5

SUMMARY OF RADAR ALTIMETER CHARACTERISTICS

PRIMARY ANTENNA TYPE PATTERN GAIN	 T-FED SLOT 90° ROLL x 160° PITCH 3.0 dB
SECONDARY ANTENNA TYPE PATTERN GAIN	 CROSSED SLOT 120 x 120 DEG 3.0 dB
TRANSMITTER TYPE FREQUENCY PRF PULSE WIDTH PEAK POWER	 TRIODE 1.0 GHz 500 Hz 5 μSEC 500 W
RECEIVER TYPE IF BANDWIDTH NOISE FIGURE	 SUPERHET, RF AMP, LEADING EDGE TRACK 200 KHz 5 dB
SYSTEM LOSSES (TWO WAY)	9 dB
PERFORMANCE MAXIMUM ALTITUDE MINIMUM ALTITUDE ACCURACY (3σ) ACQUISITION TIME DETECTION PROBABILITY FALSE ALARM TIME	 210,000 FT 5,000 FT ± 0.16% ± 630 FT 2 SEC (SINGLE SWEEP) 0.95 (SINGLE SWEEP) 1 HR
RELIABILITY	0.9912
SIZE, WEIGHT, POWER PRIMARY ANTENNA SIZE SECONDARY ANTENNA SIZE PRIMARY ANTENNA WEIGHT SECONDARY ANTENNA WEIGHT ELECTRONICS SIZE ELECTRONICS WEIGHT PRIMARY POWER	 2 x 2 x 7 IN. 2 x 7 x 7 IN. 2.2 LB 3.0 LB 400 CU.IN. 13.3 LB 28 W

circuit. The RF energy is transmitted through the duplexer and RF switch to either the primary antenna, secondary antenna, or dummy load. (The dummy load is used for checkout after canister enclosure.) A diode gate isolates the receiver during the transmit period. The return signal is amplified at RF, converted to IF, amplified at IF with AGC, envlope detected, shaped to accentuate the leading edge, and then applied to the tracking section. The tracker positions the range gate on the detected video pulse. Elapsed time measurement (transmit-to-receive) yields the altitude estimate. Acquisition is performed by programming the range gate position from maximum to minimum delay.

During the descent and entry phase the capsule is held in a preferred roll position, permitting a single T-fed slot antenna to be used. This single slot provides 90 degree roll x 160 degree pitch coverage. After aeroshell separation and during parachute descent a single crossed slot antenna (with in-phase feeding to yield linear polarization) is used to provide 120 x 120 degree coverage about the roll axis.

Physical Description - The 18.5 lb radar altimeter system consists of the electronics assembly, primary antenna, and secondary antenna.

The electronics package size, weight, and primary power requirements are 400 cu in, 13.3 lbs., and 28 w, respectively. The electronics assembly contains the RF parts section (antenna switch, isolator, circulator, and dummy load), the receiver section (diode switch, RF amplifier, local oscillators- and mixer), the IF section (IF amplifier with AGC circuitry, envelope detector, and pulse shaping network), the transmitter section (triode cavity oscillator, modulator, and high voltage power supply), the tracking and timing sections (integrated circuits and discrete parts), and the main power supply. The electronics assembly is located on the lander.

The primary antenna (single T-fed slot) is 7 x 2 x 2 in., and is mounted on the aeroshell behind an RF window. The slot is located at the preferred roll position with the seven inch dimension perpendicular to the capsule roll axis. The antenna weight is estimated to be 2.2 lb, exclusive of the window and mounting structure. The secondary antenna (cavity backed, crossed slot) is located on the lower surface of the lander. The estimated size and weight requirements are 7 x 7 x 2 in. and 3 lb, respectively.

Operational Description - Prior to deorbit, after the sterilization canister has been released, the radar altimeter is energized for self-testing. After deorbit, the altimeter is again engergized and begins transmitting via the primary antenna before an altitude of 600 000 ft is reached. Acquisition normally occurs prior to 200 000 ft altitude. At 100 000 ft a marking signal is generated to apply primary power to the landing radar. At a pre-set altitude, a marking signal is supplied to initiate parachute deployment and aeroshell release.

One second prior to aeroshell release, the sequencing function provides a "modulator holdoff/antenna select" signal, which disables the transmitter during aeroshell release and antenna switching. Four to six seconds after separation, the altimeter begins transmitting via the secondary antenna, reacquires the surface, and establishes the altitude track. To discriminate against near returns from the aeroshell, 5 microsecond receiver blanking is used. At a pre-set altitude, a marking signal is generated to initiate parachute release and terminal propulsion engine ignition. Altimeter operation is terminated after landing radar acquisition.

Performance - The performance characteristics are summarized in Table 4.3.2.4-5. The basis for the accuracy estimate is given in Section 3.2.4.3. The primary error contributors are the fluctuating error due to surface roughness and the surface state altitude bias error. The indicated single sweep detection probability of 0.95 (corresponding to +3dB per pulse S/N at 210 000 ft) results in a cumulative detection probability of greater than 0.9985 at 198 000 ft altitude.

Development Status - The radar altimeter system is based on existing pulse altimetry techniques. No long lead time or state-of-the-art items are required to design and fabricate this system. Design areas requiring particular attention are:

a) Electrical Breakdown - The basic design must minimize potential breakdown problems. Anti-breakdown insulation materials should be investigated and tested to achieve reliable protection at minimum weight for the temperature and pressure extremes expected.

b) Sterilization - The altimeter design must incorporate sterilizable components and materials. Component, subassembly, and complete system testing will be required.

References

4.3.2.4-1 NASA-CR-89678; Voyager Capsule Phase B Report; Alternatives, Analyses, Selection; Vol. II; Part B-4; McDonnell-Douglas Corp.

4.3.2.5 Sequencer system. - The sequencer system elements provide the means for accomplishing, independent of earth commands, the time-based events for:

a) Equipment checkout prior to capsule separation and mission commitment.

b) Control of the separation, deorbit, entry and landing systems.

c) Control of the surface payload equipment for mission duration of a minimum of 90 days.

The memory technique and the hardware integration approach selected for these are discussed in Section 3.2.5.

4.3.2.5.1 Design requirements: The primary sequencer system design requirements and design goals are:

a) To provide precise time lapse countdown from an externally originating start signal or a mission "mark".

b) To provide a system design that has minimum size, weight, and power requirements.

c) To provide reference frequencies to other using subsystems.

d) To provide alterable memory for mission updates.

e) To provide built-in self-test capability after enclosure in the sterilization canister.

The first design requirement results from a need for precise circuit closure timing, sometimes referenced from a mission "mark" with unpredictable time occurence (i.e., sensing 0.05 g at atmospheric entry, passage of 23 000 ft altitude, etc.). This capability requires a special unit interface design approach and internal program techniques. The units associated with event sequencing for each mission phase above meet these design requirements and goals.

4.3.2.5.2 System description: A functional block diagram of the 10.2 lb sequencer system is shown in Figure 4.3.2.5-1, and a summary of characteristics is given in Table 4.3.2.5-1. The integration of all pre-landing sequencing into the computer, which must be available for guidance and control computation purposes, provides a minimum hardware weight and the most simple interface design concept

FUNCTIONAL BLOCK DIAGRAM
SEQUENCER SYSTEM

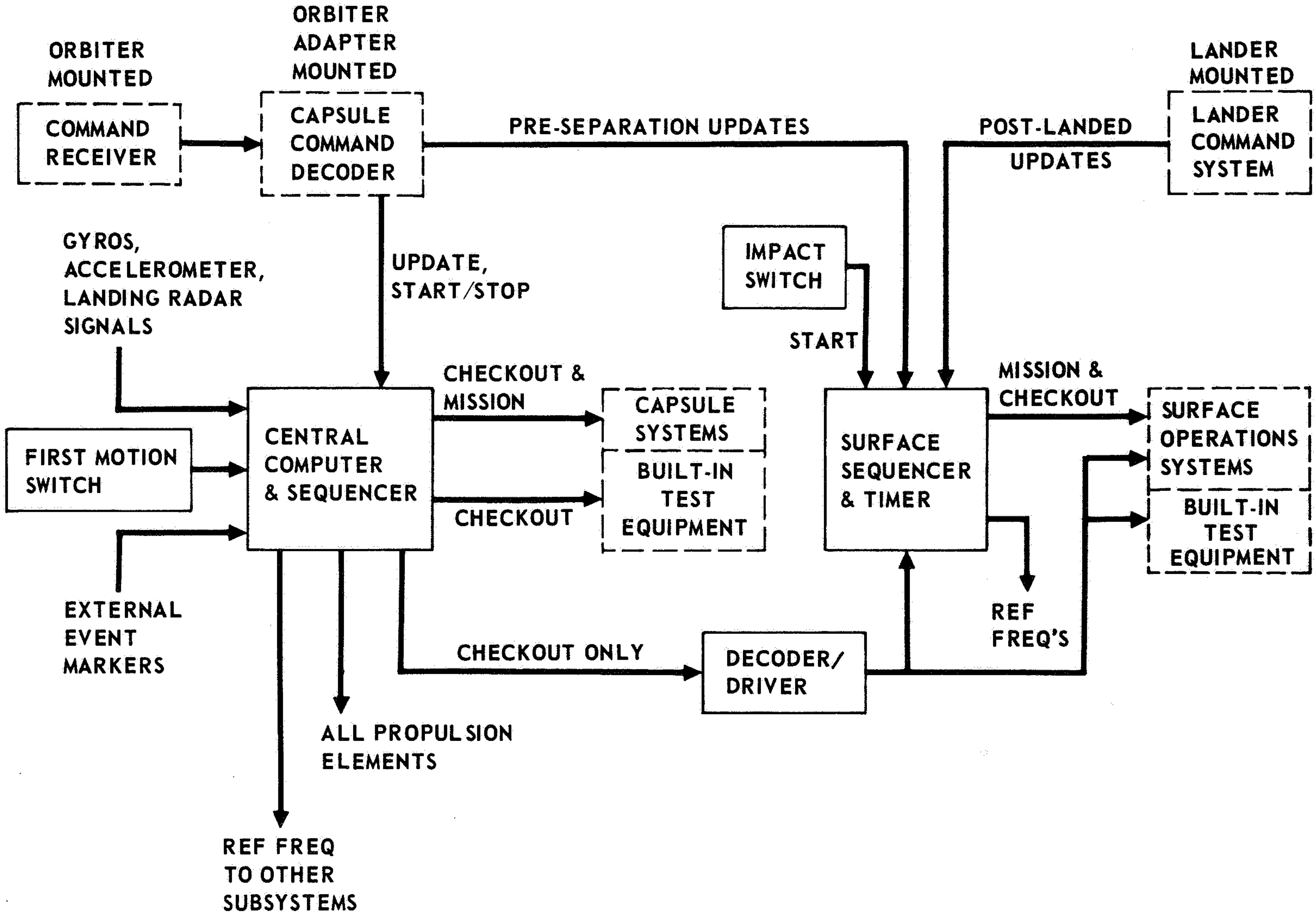


FIGURE 4.3.2.5–1

TABLE 4.3.2.5–1

SUMMARY OF SEQUENCER SYSTEM CHARACTERISTICS

CENTRAL COMPUTER AND SEQUENCER*	• SELECTABLE 50 MSEC OR 1 SEC TIME-WORD RESOLUTION • 100 PPM (3σ) CLOCK FREQUENCE STABILITY ($\pm$3.6 SEC ACCURACY IN 10 HOURS) • 555 CU IN, 20.1 LB, 34 WATTS MAX (10% POWER AND 28% SIZE AND WEIGHT DUE TO CHECKOUT AND SEQUENCING FUNCTIONS) • ALL EVENT TIME-WORDS ARE UPDATEABLE • 128 OUTPUTS FOR EACH OF CHECKOUT AND FLIGHT SEQUENCING (ALLOWS 20% GROWTH CAPABILITY)
SURFACE SEQUENCER AND TIMER	• 1 SEC TIME-WORD RESOLUTION • 100 PPM (3σ) CLOCK FREQUENCY STABILITY ($\pm$8.9) ACCURACY PER 24.6 HRS) • 240 CU IN, 8.5 LB, 10 WATTS ASYNCHRONOUS SEQUENCING 2 WATTS REPETITIVE CYCLING • ALL EVENT TIMES UPDATEABLE BEFORE AND AFTER LANDING • 128 OUTPUT CIRCUIT CAPABILITY
DECODER/DRIVER	• 20 CU IN., 1 LB, 1 WATT (ONLY USED FOR CHECKOUT TESTING)
FIRST MOTION SWITCH	• 0.2 LB
IMPACT SWITCH	• 0.5 LB

*THIS EQUIPMENT WEIGHT IS CHARGED TO THE GUIDANCE AND CONTROL SYSTEM.

(See Section 3.2.5). The separate unit to control surface operations minimizes power consumption and thermoregulation requirements for clock stability.

The Central Computer and Sequencer (CC&S) is a general purpose digital computer with a central processor, core memory, control circuitry, and input/output section (See Section 3.2.4). The memory unit employs a data word of 20 bits and has a capacity of 4096 words (512 of which are updatable). Only 11% of the stored data, 10% of the operational power, and 28% of the size and weight of the CC&S are attributable to the checkout testing and flight sequencing functions.

During checkout testing, the decoder driver decodes all CC&S test commands and selects an existing Surface Sequencer and Timer (SS&T) output driver when possible. This technique eliminates duplication of approximately 50% of the SS&T normal mission ouput drivers to perform the checkout function.

The first motion switch is an electromechanical switch that verifies orbiter/capsule separation and signals the CC&S to continue the landing flight sequence. Without receipt of this signal, the CC&S inhibits all further sequencing, to prevent ignition of the capsule deorbit motor while it is still attached to the orbiter.

The sequencer and timer for surface operation control (SS&T) is a core memory device with two major program control branches. One executive control routine provides all the one-time-only and other asynchronous event timing during the equipment activation, data transmission, and command transmission periods. During long term data gathering periods, a typical experiment sequence of 2 minutes on, followed by 13 minutes off for the same instrument and data storage equipment, is repeated. A special repeat cycle timer mode allows shutoff of the main memory, reducing power consumption by 80%.

The impact sensor is an acceleration sensing devive on the lander which senses surface touchdown and starts the SS&T for controlling the landed mission. A backup starting command is provided by the CC&S.

Both the CC&S and SS&T have a built-in self-test capability to permit system test after enclosure in the sterilization canister. The units have "fast time" and "safe test" design features, to allow running through sequence portions faster than real time without actuating mission equipment (i.e., pyrotechnics). Both units have an update capability before capsule separation, and the SS&T can be command updated after landing.

4.3.2.5.3 Operational description: Approximately 24 hours prior to capsule/orbiter separation, the CC&S activates and controls systems and their built-in test equipment (to simulate mission inputs where practical) according to the stored memory sequence. The system responses to the test commands are telemetered to earth for mission readiness evaluation. Earth commands for a test sequence recall or CC&S and SS&T memory update are made as directed by test data.

At 1 hour prior to separation, the orbiter commands the CC&S to start a separation sequence countdown. The CC&S controls the equipment start-up required and issues the separation command. The first motion switch detects the actual physical separation and notifies the CC&S to initiate the first post-separation countdown sequence.

The post-separation to surface touchdown period consists of five different sequence groups. At each consecutive mission "mark" the last sequence group data is ignored and the new sequence group time-words countdown is started. For example, at the radar altimeter "mark" of 23 000 feet altitude all prior sequence words are branched around in the computer program and the aerodecelerator deployment phase words are called into the program.

The SS&T is started by the impact switch at surface touchdown. Its equipment control actions and timeline are fixed by the memory data provided prior to separation. The mission is started by a series of one-time-only events for initial equipment activation and data acquisition. The long term pre-programmed sequence provides for a once daily communication period followed by a repetitive cycle of a certain on-off duty cycle for a particular experiment complement. The period between communication intervals is provided by counting the number of repeat cycles performed.

4.3.2.5.4 Performance: The timing accuracies indicated in Table 4.3.2.5-1 satisfy all mission design goals and impose no stringent clock specifications. The possible SS&T reference time error growth of $\pm$ 8.9 seconds per Martian day will require a clock update periodically, due to the long mission duration.

The time-word resolutions are based on mission requirements. The 50 msec quantization time of the CC&S is provided in order to time the deorbit and entry attitude maneuver command signals accurately.

4.3.2.5.5 Development status: The CC&S and SS&T can be mechanized from existing components and technology. Similar magnetic core systems have been test proven to withstand the sterilization environment. The magnetic core memory technology has been space qualified in numerous previous applications.

4.3.2.6 Thermal control. - The thermal control system maintains capsule and lander temperatures within their allowable ranges throughout all mission phases. During flight the lander is provided with a controlled temperature environment by the provisions for capsule thermal control. When exposed to the Martian surface environments the lander operates independently from the separated aeroshell. The primary results of the point design study are definition of thermal control requirements, establishment of configurations, and generation of sizing and performance data.

4.3.2.6.1 Requirements and constraints: Because the environments, requirements and constraints of the flight phase and the landed phase are appreciably different, they will be discussed separately.

Flight Phase - Equipment located within the lander is limited to a temperature range of 20 to 70°F. Several components located outside the lander, such as propulsion equipment, require active temperature control as the capsule cools during its flight away from the sun toward Mars. Electrical heaters will be required, using power drawn from the solar panels on the orbiter. The amount of power needed depends on the solar distance, and the performance of multilayer insulation provided on the forward portion of the sterilization canister. The isotope heaters used for post-landing thermal control will also reduce flight phase electrical heating requirements. The canister must be retained during Mars orbit to avoid losing the insulation. The capsule environment changes during orbital descent as the capsule becomes directly exposed to the combined solar/deep space environment. An important consideration during this period is to avoid overcooling the ablative heat shield. During the Voyager Phase B study a minimum design temperature of -150°F was selected, to avoid possible low temperature degradation. However, preliminary testing of McDonnell ablator material has demonstrated integrity to -300°F.

Landed Phase - The major requirement of the thermal control design is to provide sufficient heat retention by insulation and internal heating to prevent overcooling equipment during cyclic nighttime or continuous cold operation, while concurrently avoiding overheating during Mars cyclic daytime operations. The temperature limits of operating equipment are divided into two parts: 1) equipment requiring 0 to 100°F, and 2) equipment requiring 50 to 125°F. These ranges apply to equipment installed inside the lander. Other experimental equipment located outside the lander will experience the full range of Mars surface ambient conditions, and these must be individually temperature controlled.

The equipment power profile consists of a continuous power level of 10 watts, with peaks to 56 watts during relay transmission periods. The post-landing environment consists of repeated nominal cyclic days with day and night extremes of 120 and -150°F respectively, for both ground and atmosphere, and a continuous cold (-190°F) day. A sand and dust environment that will degrade thermal control coatings has been assumed. Winds, which affect lander heating and cooling, have also been assumed. Additional details on the Mars surface environment, plus those of the flight phases, are presented in Section 3.2.6.

4.3.2.6.2 System description: Major elements of the thermal control system are insulation blankets and panels, electrical and radioisotope heaters, and phase change of material. In addition, a thermal curtain is provided over the base of the aeroshell to retain heat during orbital descent, deflect propellant reaction products, and reduce base heating to aeroshell mounted equipment. Optical coatings (paints or tapes) will be used on surfaces that require specific optical properties.

Flight Phase Equipment - The major thermal control components for the flight phase have been defined for a flight orientation in which the capsule is partially shaded by the solar panels on the orbiter.

o Multilayer Insulation Blanket - The multilayer insulation blanket is located on the outside of the forward portion of the sterilization canister. It provides the primary thermal resistance between the capsule and the deep space environment. The blanket consists of layers of plastic film with a highly reflective coating on each layer. As such, it acts as a series of radiation heat transfer barriers. This is the most efficient type of insulation for covering large surface areas in a vacuum environment.

o Electrical Resistance Heaters and Thermostat Control Devices - Heat must be generated within the capsule to replace that which is lost to space if an acceptable equilibrium temperature level is to be maintained. The heat will be generated with electrical resistance heaters powered by solar panels on the orbiter. These heaters will be placed on components that require a higher minimum temperature than the average capsule environment. Thermostat control devices will be used with heaters that have a capacity greater than 3 watts, to conserve power and avoid overheating.

o Insulation Packages for Individual Components - The components that are provided with heaters will also be insulated, to minimize the heat loss and conserve heater power. The proposed insulation is a fiberglass type that can be efficiently fabricated to fit the individual components. Insulation of individual components is also necessary, to preclude excessive cooldown during periods when power from the orbiter is not available for heaters (e.g., during trajectory corrections).

o Thermal Curtain - The thermal curtain is required to protect components and structure within the aeroshell from aerodynamic heating during atmospheric entry. It also prevents damage to the lander thermal coatings during deorbit motor firing and reduces heat loss from the rear of the aeroshell during orbital descent.

o Thermal Control Coatings - Preparation of the surface of most aeroshell mounted equipment and structure is necessary to distribute the internal heat properly. The most critical surface is the exterior of the aft portion of the sterilization canister. The optical properties, solar absorptance and infrared emittance, of this surface determine the overall temperature levels of the capsule and their change as the solar distance increases from Earth to Mars.

Landed Phase Equipment - The thermal control equipment in the lander must be sufficient to maintain equipment temperature limits and also be compatible with the flight phase environments.

o Insulation - Thermal insulation that completely surrounds the lander is provided to protect equipment from the low Martian nighttime temperatures.

This insulation along with internal heat generation provides a temperature range of 0 to 100°F for lander equipment. Within this volume a smaller insulated compartment is provided which maintains the battery, transmitter, and sequencer from 50° to 125°F. This arrangement results in less total insulation weight than would be required if all equipment were maintained between 50° and 125°F.

o Isotope Heaters - Heat lost through the insulation is replaced by isotope heaters. These are completely passive and operate throughout the mission by radioactive decay of isotope material, probably promethium 147 or plutonium 238.

o Electrical Heaters - Electrical heaters with thermostat control are provided within the battery compartment to supply energy as needed to maintain the 50°F minimum compartment temperature.

o Phase Change Materials - Self-contained phase change material is provided to increase the thermal inertia of the lander. This approach effectively dampens the temperature variations of the lander in the cyclical external environment. The phase change material consists of a hydrocarbon wax which melts at 117°F and internal honeycomb to increase the heat transfer performance.

o Thermal Control Coatings - The external surfaces of the lander are subject to the combined solar heating and Martian surface environment. Thermal coatings are required that have low solar absorptance and are sand and dust resistant, to avoid excessive thermal performance degradation. The thermal analysis was performed assuming that the coatings were degraded to values experienced in McDonnell tests, i.e., with solar absorptance of 0.5 and emittance of 0.9.

4.3.2.6.3 Physical description: The selected thermal control design is defined in terms of flight phase and landed phase weights. The sum of these weights is the total thermal control weight allocation for the mission. These weights were based on the parametric studies described in Section 3.2.6.

Flight Phase Equipment - The selected flight phase configuration results in use of the thermal control components described below:

Flight Phase Thermal Control Components

Component	Weight (lb)	Description
Canister-Mounted Equipment	29.1	Multilayer insulation blanket (1/2 in. thick), thermal coatings
Aeroshell-Mounted Equipment	4.5	Equipment insulation, thermal coatings, heaters
Thermal Curtain	-	Silica Cloth (see Section 4.3.2.9)
Lander-Mounted Equipment	26.4	Equipment and tankage insulation, heaters
Total	60.0 lb	

Landed Equipment - The selected Martian Soft Lander is shown schematically in Figure 4.3.2.6-1. Included in this sketch are the various thermal control devices located within the two compartment lander design. A weight breakdown for the thermal control system is given below.

Lander Thermal Control Components

Component	Weight (lb)	Description
Outer insulation	27.5	3" thick, 4 lb/ft^3
Isotopes in outer compartment	2.5	12.5 watts
Battery compartment insulation	1.8	0.4" thick, 4 lb/ft^3
Isotopes in battery compartment	10.2	51 watts
Phase change material in battery compartment	7.6	hydrocarbon wax
Total	49.6 lb	

(This does not include 17.0 lb of batteries required to supply heater power.)

THERMAL CONTROL SCHEMATIC – DUAL COMPARTMENT LANDER

CONCEPT III

- **EXTERNAL ENVIRONMENT –
  90 CYCLIC DAYS (–150 TO + 120°F)
  WITH ONE COLD DAY (–190°F)**

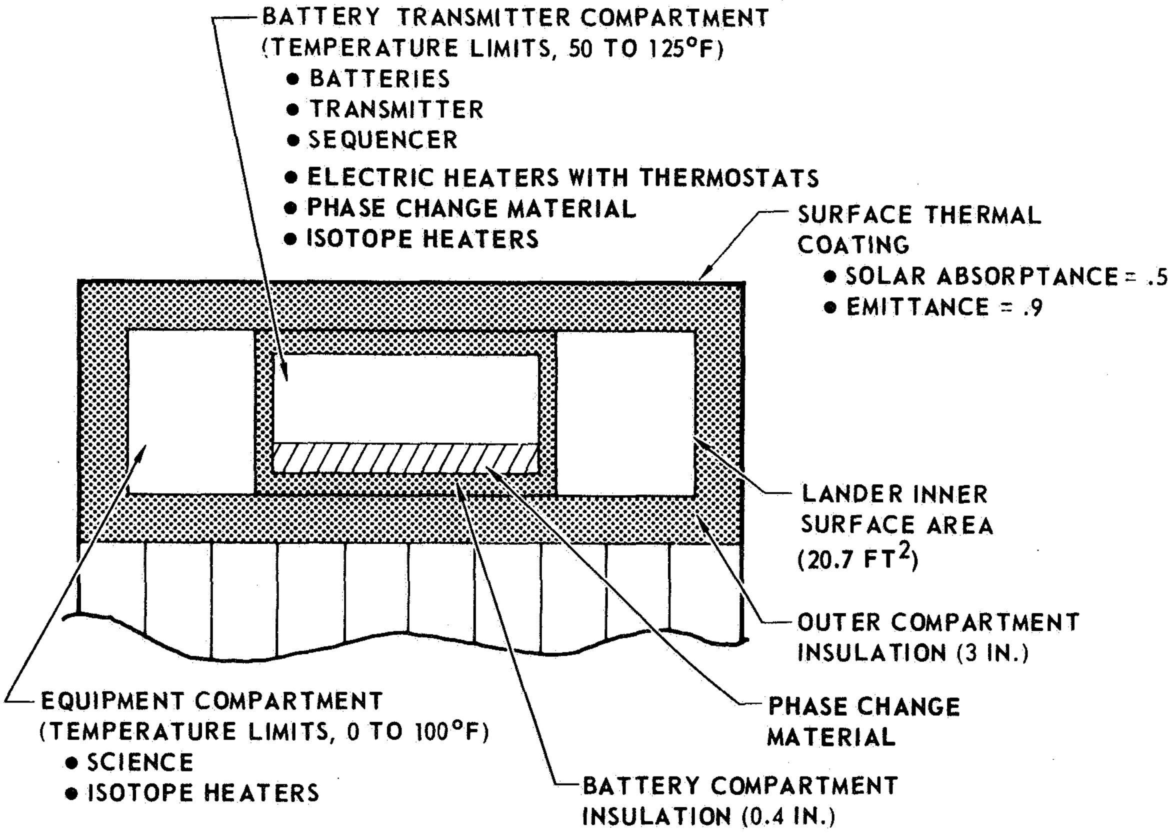


FIGURE 4.3.2.6–1

4.3.2.6.4 Operational description: The thermal control system operates throughout the mission to maintain equipment and structural temperatures using combinations of constant power level isotope heaters and thermostatically controlled electrical heaters. Heater power requirements are reduced by use of thermal coatings and insulation.

Flight Phase - In the vicinity of Earth, solar heating of the capsule is sufficient to avoid use of heaters. However, as the solar distance gradually increase the capsule cools until temperatures reach the thermostat control points. Thereafter, the electrical heaters maintain controlled temperatures.

The sterilization canister is retained throughout the Mars orbit period to avoid excessive cooling of equipment and increased heater power.

After separation, and during orbital descent, the thermal inertia of the equipment, the resistance to heat loss by the insulation, and the supply of internally generated heat results in the avoidance of need for heater power.

Landed Phase - The post-landing thermal control system is capable of maintaining both lander compartments above their respective lower temperature limits for extended cyclic days of operation, with provisions to encounter one continuous cold day.

For the extended cyclic days, sufficient constant isotope heating and power profile heating maintain the two compartments well above their minimum operating temperatures. During daytime operation, lander temperatures rise and the battery compartment approaches its maximum temperature limit. Subsequent melting of phase change material in this compartment prohibits excessive temperature rise. The utilization of constant isotope heating reduces the required battery weight for continuous cold operation. During the nighttime the high thermal inertia and constant supply of isotope and power profile heating prevents excessive temperature reduction in both compartments, and no controllable heating is required.

For the one continuous cold day, the additional heating required to maintain the battery compartment above its lower temperature limit is supplied by thermostatically controlled electrical heaters. The heating provided by isotopes equipment, power profiles, and heat losses from the battery compartment, enables the equipment compartment to remain above its minimum operating temperature without the need for additional electrical heating.

4.3.2.6.5 Performance characteristics: The lander thermal performance is characterized by the temperature variations within the two compartments for extended cyclic temperature environment operation and for one continuous cold day. During the cyclic days of operation maximum and minimum battery compartment temperature are 125°F and 65°F, respectively. Similarly, the equipment compartment temperature varies between 100°F to 20°F. These are temperatures obtained after several days operation in identical cyclic environments. A gradual daily increase in stored heat occurs for many days after landing because of repeated melting and partial freezing of the phase change material.

During a continuous cold day both compartments require continuous heater power, except for small transient carryovers from the cyclic day. For the most part the battery compartment and the equipment compartment remain at their lower temperature limits, 50°F and 0°F, respectively.

4.3.2.6.6 Development status: The major thermal control development problem is the lander insulation. Its performance in the Martian surface environment after exposure to the mission environments of sterilization, launch vibration, long term vacuum, entry, and landing has not been evaluated. Particularly uncertain is the effect of insulation installation and attachment techniques on overall thermal performance.

Long term space exposure effects on the aft canister thermal coating are a potential problem. If the degradation characterized by increased solar absorptance is neither too rapid nor too large, the absorptance change is beneficial to capsule thermal control since temperatures will not decrease as much during the flight from Earth to Mars.

A second thermal coating problem occurs after landing when the lander exterior coatings are exposed to the Martian sand and dust environment. Preliminary McDonnell tests have indicated that solar absorptance can increase by a factor of two or more after exposure, and that extended exposure can result in complete removal of the coating material. Several coating materials, specifically flame sprayed aluminum oxide, white pigmented silicone elastomer and white porcelain enamel have demonstrated resistance to the environment with respect to both retention of optical properties and erosion resistance.

4.3.2.7 Propulsion. - The propulsion functions required by the Mars capsule are accomplished with three separate systems: a solid propellant rocket motor for deorbit, a cold gas system for attitude control and a throttleable, monopropellant system for terminal deceleration. The description, usage and development status of these systems are discussed in this section.

4.3.2.7.1 Deorbit propulsion system:

System Description - The deorbit propulsion system is a spherical, solid propellant rocket motor, containing provision for thrust termination prior to propellant burnout. The motor configuration is shown in Figure 4.3.2.7-1 and is functionally described in Figure 4.3.2.7-2. The motor consists of a titanium case; an igniter assembly (consisting of an igniter and a safe and arm device); a partially submerged, high expansion ratio nozzle; and an internal-perforated, case bonded propellant grain. The propellant is a saturated, hydroxy-terminated polybutadiene/aluminum/ammonium perchlorate formulation providing good thermal stability and adequate physical properties at the required sterilization temperature.

A thrust-termination mechanism holds the nozzle to the motorcase. The mechanism consists of a split clamp ring and a pyrotechnically-actuated retention bolt.

The physical and performance characteristics of the rocket motor are presented in Figure 4.3.2.7-3.

Operational Description - Arming of the deorbit igniter is affected by a signal from the timer and sequencer 55 minutes before capsule-orbiter separation. Upon receipt of this signal, an electro-mechanical actuator on the igniter safe and arm (S&A) physically aligns the pyrotechnic initiators with the igniter. A position transducer on the S&A verifies that the igniter is armed. Approximately 5 seconds prior to deorbit rocket ignition, the ignition system is enabled by a command from the capsule timer and sequencer. At the command to ignite the deorbit motor, the pyrotechnic initiators ignite the pellets in the igniter which in turn ignite the igniter sustainer charge. The igniter sustainer exhausts into the motor propellant cavity, providing sufficient heat and pressure to ignite the main propellant charge.

DEORBIT MOTOR

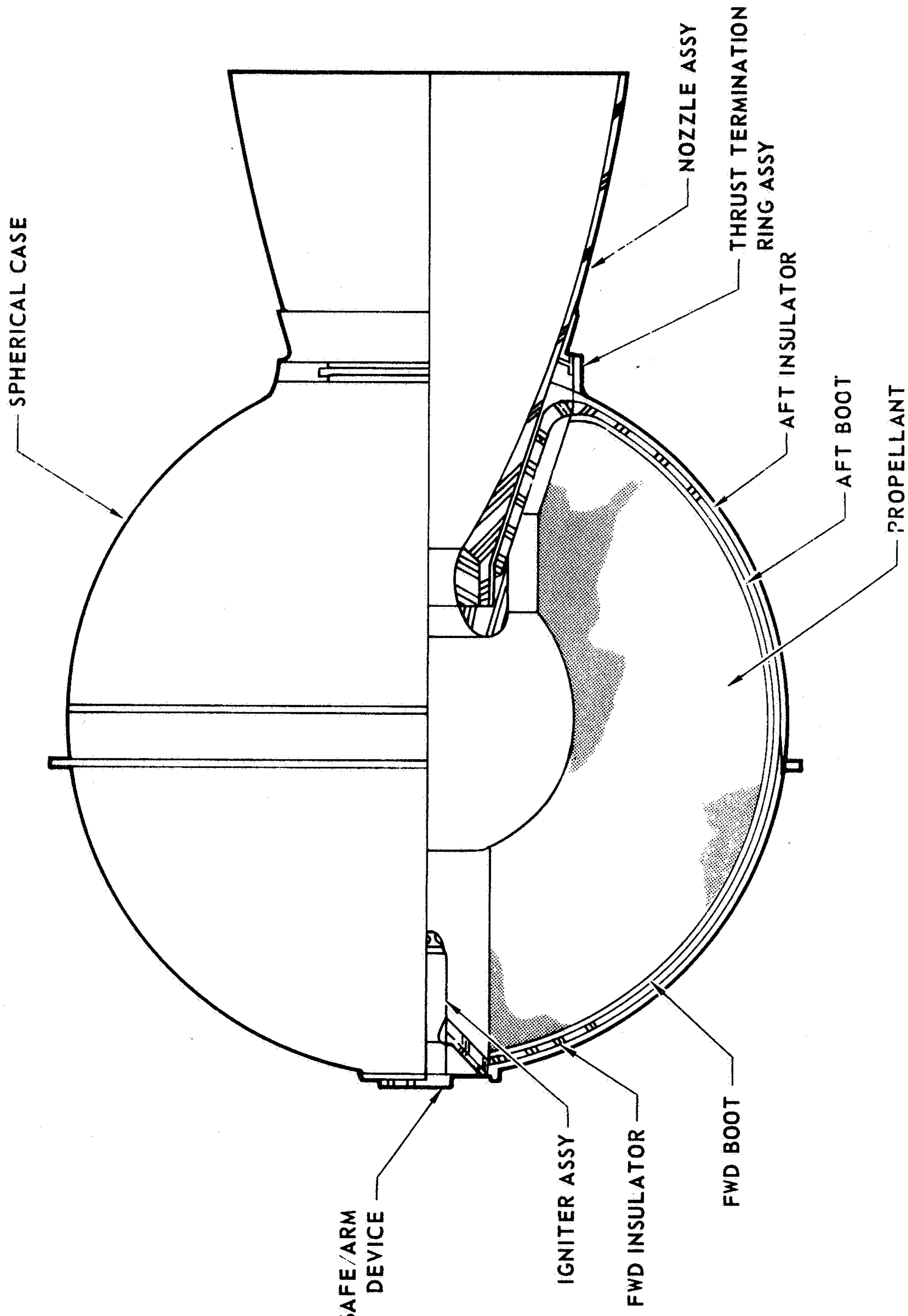


FIGURE 4.3.2.7-1

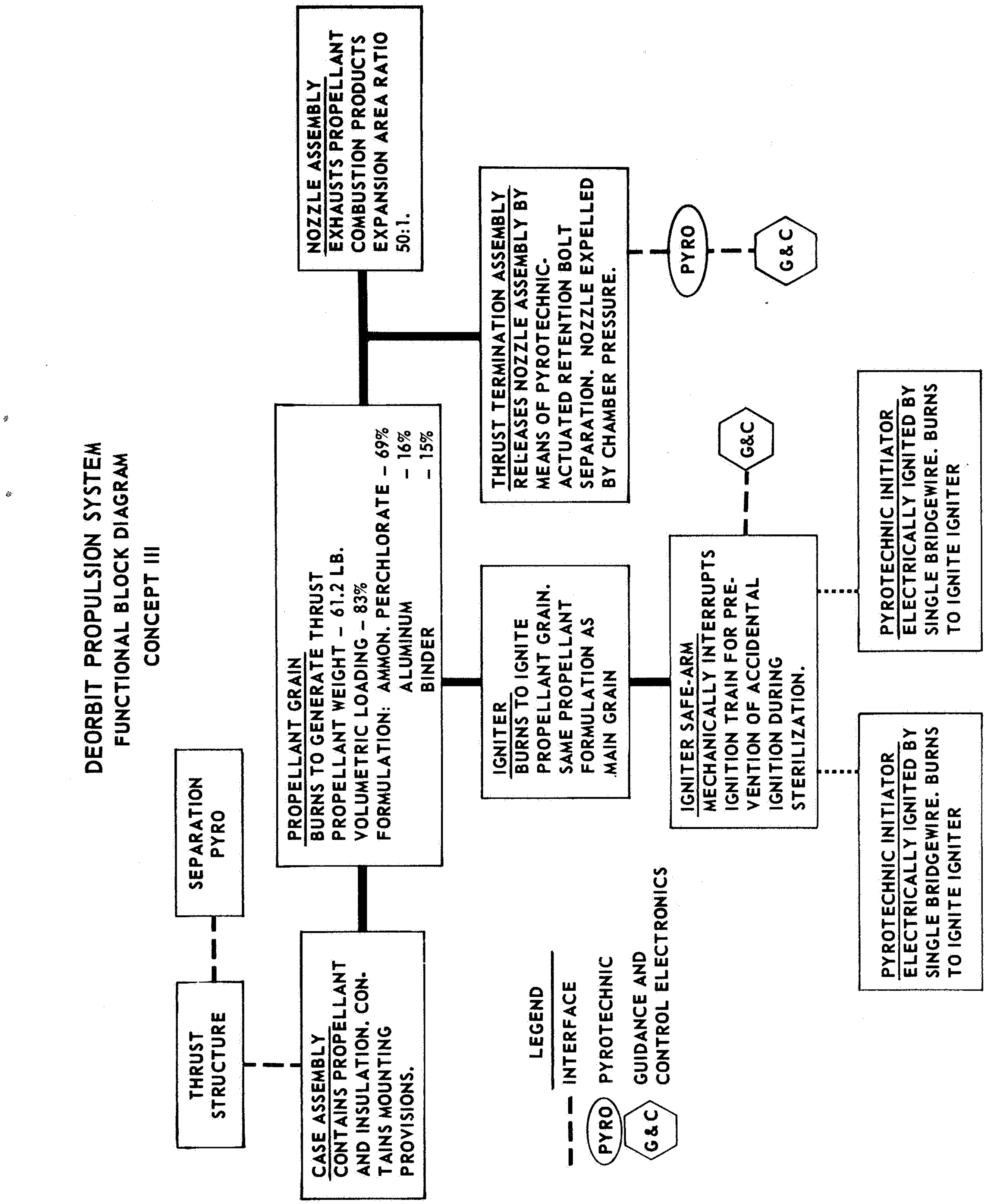


FIGURE 4.3.2.7–2

CONCEPT III

DEORBIT MOTOR CHARACTERISTICS

(DEORBIT $\Delta V = 450$ FT/SEC)

MOTOR	
LENGTH (OVERALL)	19.9 IN.
DIAMETER (SPHERICAL)	13.8 IN.
WEIGHT	
INERT	12.6 LB
LOADED	80.7 LB
CASE	
MATERIAL	6 Al–4V TITANIUM
WEIGHT	6.2 LB
INSULATION	
WEIGHT	2.5 LB
NOZZLE	
TYPE	ABLATIVE
CONFIGURATION	CONTOURED
WEIGHT	3.0 LB
EXPANSION RATIO	50
THROAT DIAMETER	1.29 IN.
EXIT DIAMETER	9.11 IN.
IGNITER	
TYPE	PYROGEN
WEIGHT	.9 LB
NO. OF SQUIBS	2
PROPELLANT	
TYPE	PB(16% Al–69% AP)
PREDICTED I_{SP} AT 600 PSI (VACUUM)	288 SEC
DENSITY	0.060 LBM IN^3
MOTOR PERFORMANCE	
AVERAGE ACCELERATION	< 1 G
BURNING TIME	13.8 SEC
AVERAGE THRUST	1420 LB
AVERAGE CHAMBER PRESSURE	600 PSIA
VACUUM TOTAL IMPULSE	19,600 LB-SEC
VOLUMETRIC LOADING	82%

FIGURE 4.3.2.7–3

Simultaneous to the command to ignite the motor, the timer and sequencer arms the pyrotechnic retention bolt on the thrust termination mechanism. Thrust termination of the deorbit motor is achieved by releasing the nozzle. By activating the contained explosive bolt, the thrust termination ring that attaches the nozzle to the motorcase is released. The sudden pressure decrease associated with releasing the nozzle terminates propellant burning. Thrust termination can be initiated at any time after ignition. A thrust spike will occur at thrust termination; however, the duration of the spike is extremely short ($\sim$20 ms). This technique of thrust termination has been used successfully in the Jupiter and Titan II vernier rocket motor applications.

Development Status - The deorbit motor as depicted in Figure 4.3.2.7-1 is a conventional configuration, containing no design features which have not been flight proven in previous space applications. The design is very similar to the Titan II vernier, Gemini retro and Surveyor retro rockets. Nevertheless, a major development effort is required to meet the sterilization requirements. Sterilization feasibility has not been adequately demonstrated on a solid propellant motor of applicable size. Current off-the-shelf propellants, liners and insulations have exhibited serious degradation when exposed to the sterilization environment. New propellant systems containing saturated binders and stabilized oxidizer appear very promising based on tests of small samples. However, the only motor of significant size to be loaded with a saturated propellant and subjected to the sterilization environment was a 50 lb_m Syncom motor, tested by JPL. The motor survived one heat cycle without noticeable adverse effects but failed in the subsequent heat cycle due to propellant-steel incompatibility. (The motorcase was unlined.) Considering the failure mode, it is significant that the motor survived the first heat cycle in good condition.

Sterilizable, high performance motor designs are believed to be achievable with these latest propellant developments, but more development work must be done, particularly on insulation materials. Plasticizers and dehydrated fillers in the insulation can degrade the bond between the propellant and insulation. Many of the methods employed to enhance the stability and physical properties of the propellant at high temperature are suitable, as well, for insulation materials. This must be accomplished, however, without serious degradation to the insulation properties.

Other available component materials such as O-rings, nozzles and igniters appear to be adequate.

4.3.2.7.2 Attitude control system:

System Description - Attitude control of the capsule is provided by a cold gas propulsion system utilizing gaseous nitrogen as the expellant. The functions performed by the system are:

a) separation of the capsule from the orbiter, and

b) attitude control and rate damping of the capsule during deorbit, descent, and atmospheric entry.

The attitude control system is presented schematically in Figure 4.3.2.7-4 and is functionally described in Figure 4.3.2.7-5. Gaseous nitrogen is stored at high pressure in a spherical pressurant tank. The nitrogen is isolated from the remainder of the system by a pyrotechnic-operated valve. Upon system activation, the isolation valve is actuated allowing the gas to flow from the storage tank to the regulator where the pressure is reduced to design operating pressure. The gas emerging from the regulator is then distributed to the eight thrust chamber assemblies installed about the periphery of the aeroshell. The regulator and thrust chamber assemblies are protected from contamination by in-line filters. Test ports are provided to allow for pre-flight testing of the system components. System condition prior to activation and during operation is monitored by temperature and pressure instrumentation. The performance and physical characteristics of the attitude control system are presented in Figure 4.1.2.7-6.

Operational Description - The attitude control system is loaded with gaseous nitrogen and sealed before terminal sterilization. Approximately 5 seconds prior to separation of the capsule from the orbiter, the attitude control system is activated by actuating the pyrotechnic isolation valve at the outlet of the nitrogen storage tank. Electrical signals to the eight TCA solenoid valves permit nitrogen flow and thrust. Four TCA's provide pitch and yaw control while the remaining four provide roll control.

ATTITUDE CONTROL SYSTEM SCHEMATIC

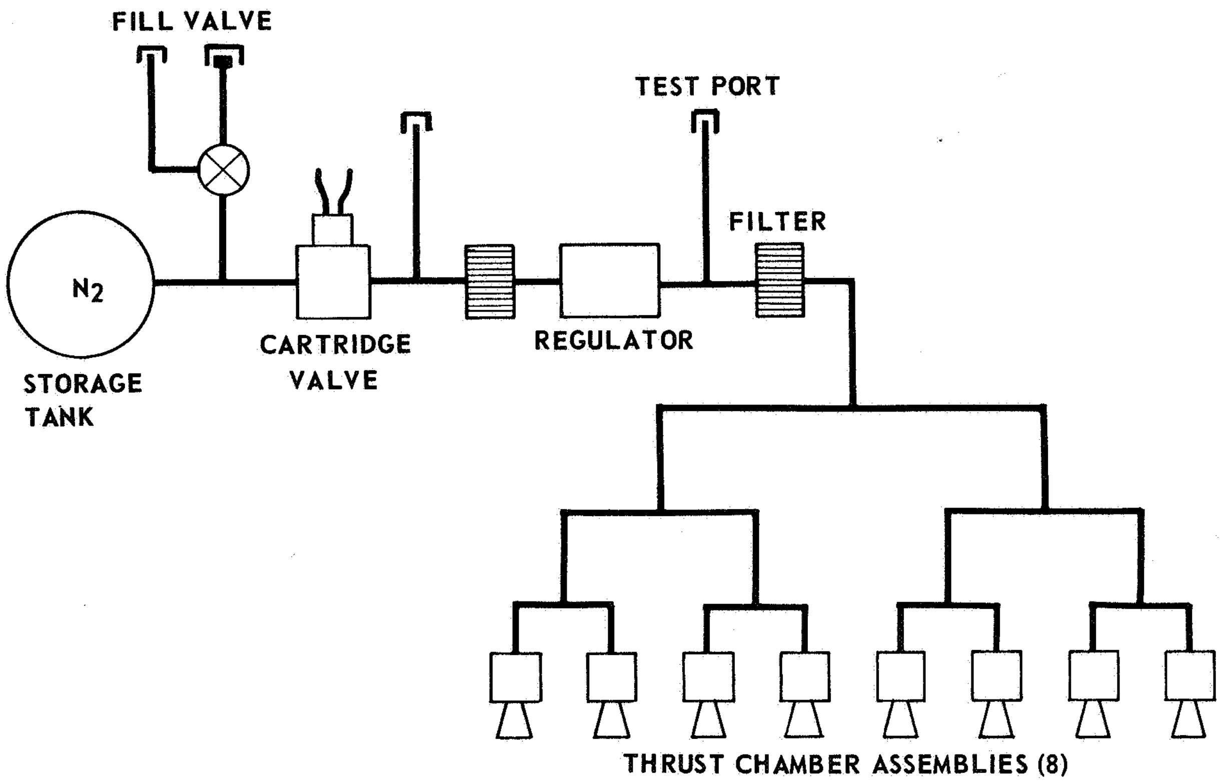


FIGURE 4.3.2.7–4

ATTITUDE CONTROL SYSTEM
FUNCTIONAL BLOCK DIAGRAM
CONCEPT III

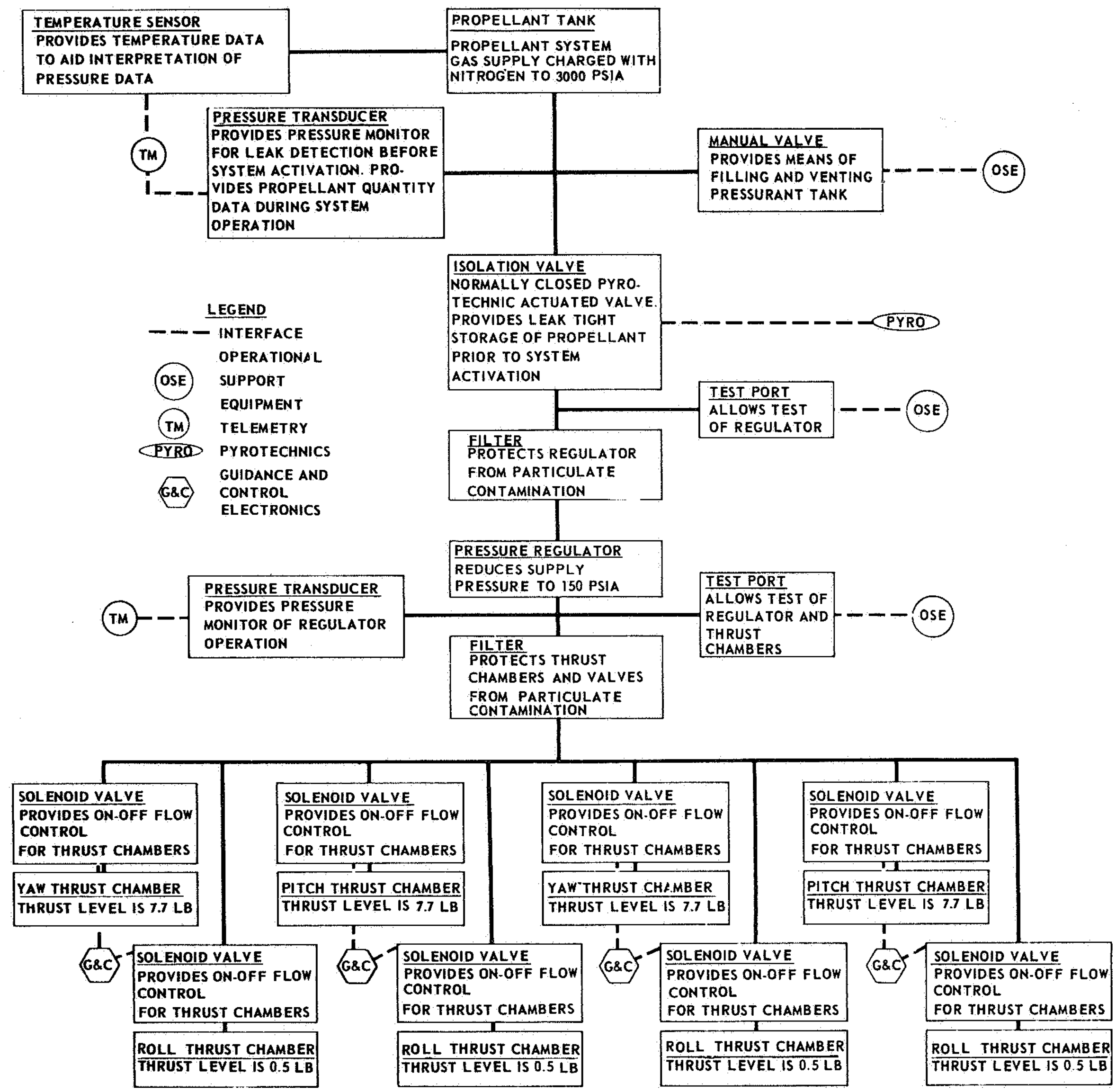


FIGURE 4.3.2.7–5

CONCEPT III
ATTITUDE CONTROL SYSTEM CHARACTERISTICS

SYSTEM

TYPE	COLD GAS
PROPELLANT	NITROGEN
PRESSURIZATION	REGULATED
TOTAL IMPULSE, LB-SEC	330
TOTAL WEIGHT, LOADED, LB	32.6
TOTAL WEIGHT, DRY, LB	28.0

THRUST CHAMBER ASSEMBLIES

TOTAL NUMBER	8
TYPE	FIXED THRUST, FIXED MOUNT

	PITCH	YAW	ROLL
NO. PER AXIS	2	2	4
NO. PER CONTROL MANEUVER	1	1	2
THRUST PER TCA, LB	7.7	7.7	0.5
SPECIFIC IMPULSE, SEC	72.5	72.5	72.5
MINIMUM IMPULSE PULSE, LB-SEC	.0666	.0666	.0045
RESPONSE, START, SIGNAL TO 90% THRUST, SEC.	.015 MAX.	.015 MAX.	.015 MAX.
RESPONSE, SHUTDOWN, SIGNAL TO 10% THRUST, SEC	.010 MAX.	.010 MAX.	.010 MAX.
AREA RATIO, ϵ	30	30	30
CHAMBER DIAMETER, IN.	.24	.24	.07
NOZZLE EXIT DIAMETER, IN.	1.73	1.73	.34

TANKAGE

MATERIAL	TITANIUM
ENVELOPE, IN	10.4 DIA

PRESSURES	CHARGE	STERILIZATION	OPERATION
STORAGE TANK, PSIA	3000	4520	3000–300
REGULATOR INLET, PSIA	14.7	20.4	3000–300
REGULATOR OUTLET, PSIA	14.7	20.4	150

FIGURE 4.3.2.7–6

The separation of the capsule from the sterilization canister is performed by firing the four pitch and yaw TCA's simultaneously to provide a separation velocity increment of 1.5 feet per second. Attitude control during this maneuver is affected by utilizing reverse guidance logic wherein the thruster(s) producing a control moment in the direction of the disturbance is momentarily shut-down. Orbit attitude control, attitude stabilization during deorbit motor firing, descent attitude control, and attitude rate damping during Mars atmospheric entry is obtained by pulsing the TCA's in response to guidance and control commands. Approximately 4 seconds after full deployment of the parachute, the aeroshell, including the attitude control system, is jettisoned.

Development Status - Extensive development of cold gas nitrogen propulsion systems has been performed. The Ranger, Mariner, Surveyor, and Lunar Orbiter vehicles have used cold gas nitrogen systems. From these and other programs, hardware adaptable to Mars capsule requirements are available. Additional effort will be required to meet the mission requirements and to develop compatibility with the sterilization environment. Sterilization of a cold gas propulsion system will affect the design of the storage tank, regulator and valves.

4.3.2.7.3 Terminal propulsion system:

System Description - The terminal propulsion system is a gas-pressurized, monopropellant hydrazine system containing three variable-thrust engines. A schematic of the system is shown in Figure 4.3.2.7-7 and is functionally described in Figure 4.3.2.7-8.

System pressurization is accomplished with high pressure helium gas which is stored in a spherical 6Al-4V titanium tank. The helium pressurant is isolated before use by a normally-closed pyrotechnic valve installed immediately downstream of the pressurant tank. A manual access valve provides a means of pressurant servicing, a regulator is provided to maintain constant propellant feed pressure. The regulator is isolated from propellant vapors during the long Earth-Mars transit time by normally-closed pyrotechnic valves. Relief valves are employed to provide system protection in the event of an overpressure situation. Burst diaphragms separate the relief valves from the pressurant system to prevent relief valve leakage at normal operating pressure.

SCHEMATIC DIAGRAM OF TERMINAL PROPULSION SYSTEM

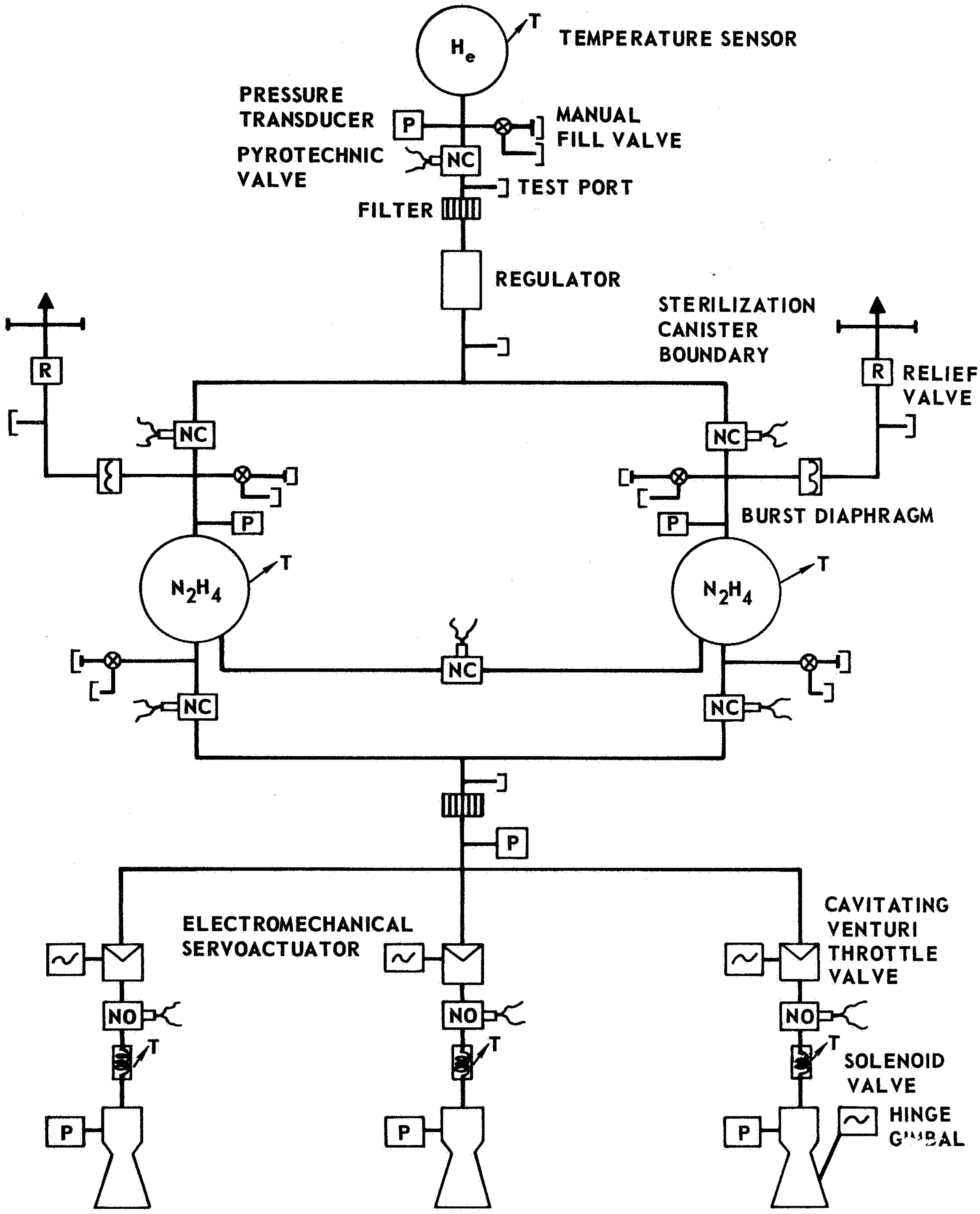


FIGURE 4.3.2.7-7

TERMINAL PROPULSION SYSTEM
FUNCTIONAL BLOCK DIAGRAM
CONCEPT III

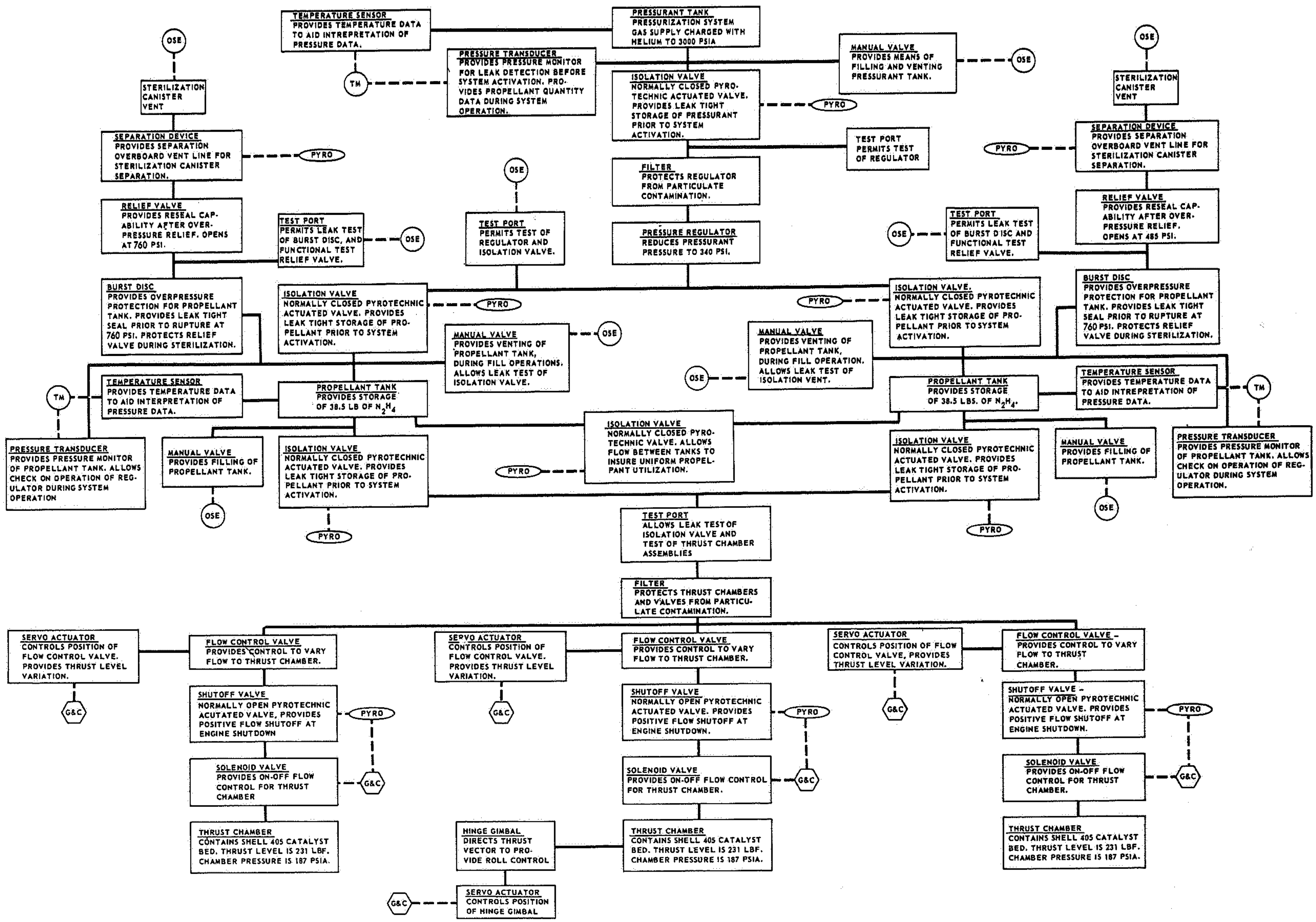


FIGURE 4.3.2.7–8

The hydrazine propellant is stored in two equal volume tanks. These tanks are spherical in shape and are fabricated from 6Al-4V titanium to minimize propellant decomposition. A large diameter transfer manifold between the tanks ensures equal propellant utilization during system operation. Propellant migration during the zero-g cruise and orbit portions of the mission is prevented by installation of a normally-closed pyrotechnic valve in the propellant transfer manifold. Manual access valves upstream and downstream of the tanks provide means of propellant servicing. The propellant is isolated before use by normally-closed pyrotechnic valves.

Critical components are protected from contamination by in-line filters. Test ports are installed throughout the system to facilitate system checkout.

Each of the three engines employs a cavitating venturi throttle valve, solenoid valve, and thrust chamber assembly (TCA). The cavitating venturi throttle valve serves two functions: first, it controls propellant flow rate to the engine, and second, it decouples the feed system from downstream pressure oscillations, thereby minimizing the possibility of overall system instability. The TCA employs a catalytic decomposition reaction chamber using Shell 405 spontaneous catalyst. A hinge-gimbal on one engine provides roll control.

The physical and performance characteristics of the terminal propulsion system are presented in Figure 4.3.2.7-9.

<u>Operational Description</u> - Operation of the TPS is initiated during the final moments of aerodeceleration (parachute or AID).

Propellant settling results from atmospheric drag-induced deceleration of the lander prior to system arming (no supplementary positive expulsion or propellant orientation devices are required). The pressurization sequence is initiated when the signal is given for deployment of the parachute or AID. At this point the isolation valve pyrotechnic circuits are armed. Following an approximate 12 second delay, the normally closed propellant tank isolation valves are fired permitting propellant to flow to the engine solenoid valves under its own vapor pressure. The pressurant system isolation valves are fired 5 seconds later to pressurize the propellant ullage. This pressurization

CONCEPT III
TERMINAL PROPULSION SYSTEM CHARACTERISTICS

DESIGN CONDITIONS	
ALTITUDE AT IGNITION, FT	6500
VELOCITY AT IGNITION, FT/SEC	376
HORIZONTAL WIND VELOCITY, FT/SEC	220
SYSTEM	
TYPE	MONOPROPELLANT
PROPELLANT	HYDRAZINE
IGNITION	CATALYST, SHELL 405
PRESSURIZATION	STORED COLD GAS, REGULATED
PRESSURANT	HELIUM
TOTAL IMPULSE, LB-SEC	21 100
TOTAL WEIGHT, LOADED, LB	229
PROPELLANT WEIGHT, LB	98
PRESSURANT WEIGHT, LB	1.41
THRUST CHAMBER ASSEMBLIES	
TOTAL NUMBER	3
TYPE	THROTTLEABLE, RADIATION COOLED
THRUST/ENGINE, LB	354 THROTTLEABLE, 354 TO 75
SPECIFIC IMPULSE (STEADY STATE, MAXIMUM THRUST) LBF-SEC/LBM	233.5
RESPONSE, MILLISEC	
0% TO 90% THRUST	200
100% TO 10% THRUST	150
10% TO 100% THRUST	150
LIFE, SEC	100
AREA RATIO, ϵ	30
ENVELOPE, IN.	
LENGTH OVERALL (EXCLUDING VALVE)	15.26
CHAMBER DIAMETER	6.44
NOZZLE EXIT DIAMETER	6.5
TANKAGE	
PROPELLANT (NUMBER)	2
MATERIAL	TITANIUM (6 Al–4V)
ENVELOPE, IN.	14.4
PRESSURANT (NUMBER)	1
MATERIAL	TITANIUM (6 Al–4V)
ENVELOPE, IN.	13.8

PRESSURE, PSIA	CHARGE	STERILIZATION	OPERATION
PRESSURANT TANK	3000 AT 70°F	4120 AT 275°F	3000–760
REGULATOR INLET	14.7	20.4	3000–760
REGULATOR OUTLET	14.7	20.4	530
PROPELLANT TANKS	14.7	60	530
COMBUSTION CHAMBER	14.7	20.4	294–60
BURST DISC RUPTURE PRESSURE		760	
RELIEF VALVE RELIEF PRESSURE		760	

FIGURE 4.3.2.7–9

procedure precludes large hydraulic surge pressures and entrainment of pressurant bubbles in the propellant feed lines.

At a prescribed range of 6500 feet, a signal from the altimeter opens the engine solenoid valves, allowing propellant flow to the TCA. The propellant decomposes upon contact with the catalyst to provide the required thrust. Electromechanical servoactuators, controlling the cavitating venturi throttle valves and engine gimbal mechanism respond to signals from the guidance electronics to vary the thrust and roll control moment for descent and attitude control.

Exhaust contamination and alteration of the surface is minimized by shutting down the engines approximately 10 feet above the surface. A backup signal is sent simultaneously to normally open valves just upstream of the solenoids to insure against propellant leakage which would interfere with post-landed operations.

Development Status - Except for the engines, the terminal propulsion system depicted in Figure 4.3.2.7-7 is within state-of-the-art technology. Additional development effort is required to meet sterilization requirements but the feasibility of heat-sterilizing liquid propellant systems has been demonstrated at JPL and Martin-Marietta. Many problems were encountered, i.e., shifts in regulated pressure, dielectric failure of solenoid valves, seal leakage, desensitized pyrotechnics and propellant decomposition. However, failure analyses, and in some cases retest, have shown that these anomalies can be circumvented by proper material selection and careful control of detail design and process procedures.

Adaptations of many existing component designs also appear feasible. For the system described herein, a Menasco tank (PN 812500-501), with modification of outlet tube stub and increased wall thickness, is suitable for pressurant storage. This tank has a minimum volume of 1275 in^3.

Pressurant flow requirements can be met with an existing regulator; Fairchild Hiller Stratos-Western, PN 385000. The regulator is a series-redundant unit weighing 3.5 lbs and is employed in the Apollo Lunar Module ascent stage. Its rated flow rate is approximately 145 SCFM, helium. Seal modifications would be required to comply with sterilization requirements; the outlet pressure rating would be increased from 181 psi to 530 psi.

No gravity feed propellant tanks of the required pressure and volume presently exist. A new propellant tank development is therefore required. The tanks will be constructed of 6Al-4V titanium to minimize propellant decomposition. Decomposition rates measured during propellant-material compatibility tests at McDonnell indicate that a 3% ullage volume at 275°F is sufficient for in-tank sterilization.

The throttleable, monopropellant hydrazine engine poses the greatest risk to system development. The major cause of problems is the fact that the most suitable catalyst found to date (Shell 405) possesses very weak structural properties, due to the necessity of providing high porosity for good performance. The catalyst particles are subject to breakup during engine firing, causing an attendant loss in catalyst material and rough engine operation. This damage can occur through a number of mechanisms including: differential thermal expansion between the catalyst and engine wall, vibration, poor assembly procedures, and excessive injection velocity.

Maintaining injector and catalyst bed pressure drops within acceptable design limits over the full throttle range will be a major development task. Control of these parameters is believed to be less critical with an injector which provides catalyst bed penetration. This technique is employed on the Walter Kidde Inc. 270 lb_f hydrazine engine. In this design, injector tube elements extend into the catalyst bed and propellant is distributed through radial holes in each element. The engine is scheduled to complete qualification testing for a classified program in November, 1968. Due to its development status, the Kidde engine was selected for this design. The required maximum thrust level is achieved by appropriate adjustment of propellant feed pressure.

4.3.2.8 Auxiliary aerodynamic decelerator system. - The 45.5 lb decelerator system consists of the following major elements:

- o Parachute
- o Riser Assembly
- o Reefing Line Cutters
- o Deployment Bag
- o Catapult Assembly

The relationship of the system components is shown in Figure 4.3.2.8-1 and a functional block diagram is shown in Figure 4.3.2.8-2.

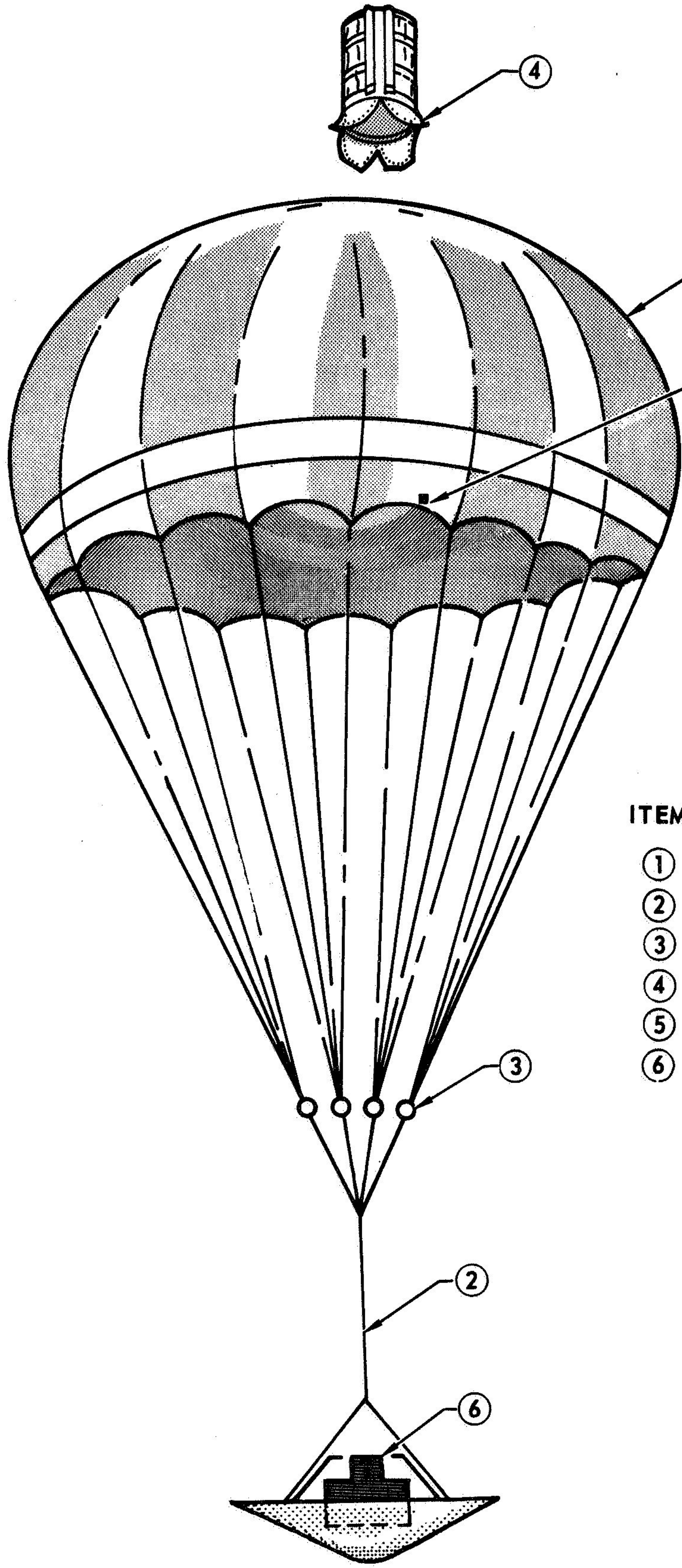
Parachute - Candidate parachute types are the ringsail, cross and disk-gap-band. For a ringsail type, the nominal parachute diameter, D_o, is 34.0 feet. The canopy material will be of the 330 Nylon or Dacron type. Preliminary analysis indicates that no reefing will be required. However, until further analysis confirms this, provisions will be made for the installation of a reefing line and reefing line cutters.

Riser Assembly - The riser assembly attaches the parachute to the capsule. It is connected to the canopy at the four suspension line connector links and to the capsule at the four structural attach points. The riser is constructed from low elongation Dacron webbing.

Reefing Line Cutters - If reefing is necessary to reduce opening shock loads on the canopy, reefing line cutters with a prescribed time delay will be used to sever the parachute reefing line. The cutters are installed in pockets sewn to the canopy skirt. Line stretch at parachute deployment pulls the firing pin to initiate the pyrotechnically actuated cutters. A typical reefing cutter which is shown in Figure 4.3.2.8-3 has an aluminum tubular body containing a spring-loaded firing pin, a time delay cartridge, and a cutter blade. A hole is provided near one end of the cutter body to insert the reefing line.

Parachute Deployment Bag - The parachute canopy, suspension lines, and a portion of the riser assembly are stowed in the parachute deployment bag

TYPICAL AERODYNAMIC DECELERATOR SYSTEM



ITEM	DESCRIPTION
1	PARACHUTE
2	RISER
3	LINK
4	DEPLOYMENT BAG
5	REEFING LINE CUTTER
6	CATAPULT ASSEMBLY

FIGURE 4.3.2.8-1

AUXILIARY AERODYNAMIC DECELERATOR SYSTEM
FUNCTIONAL BLOCK DIAGRAM

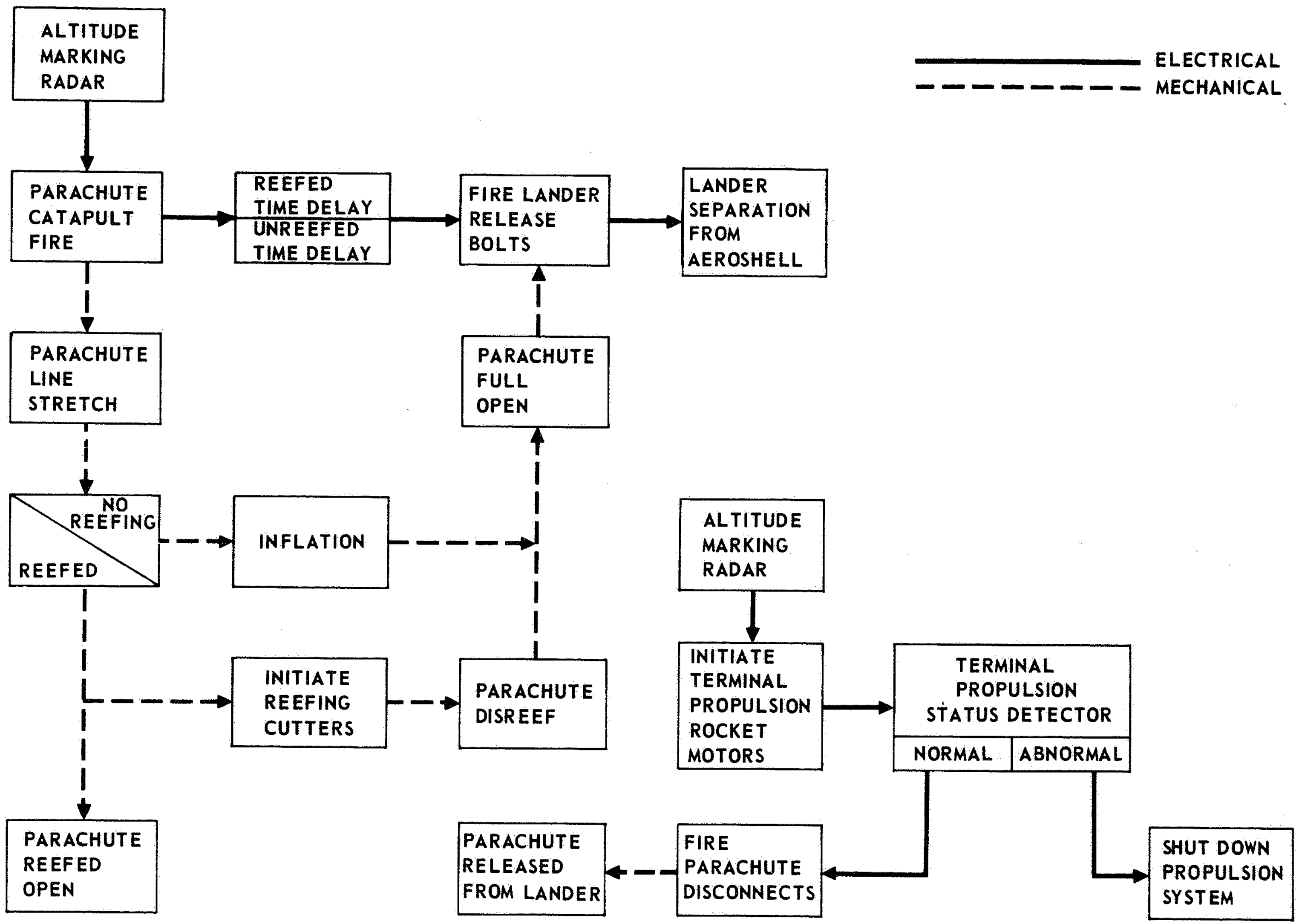


FIGURE 4.3.2.8–2

REEFING LINE CUTTER

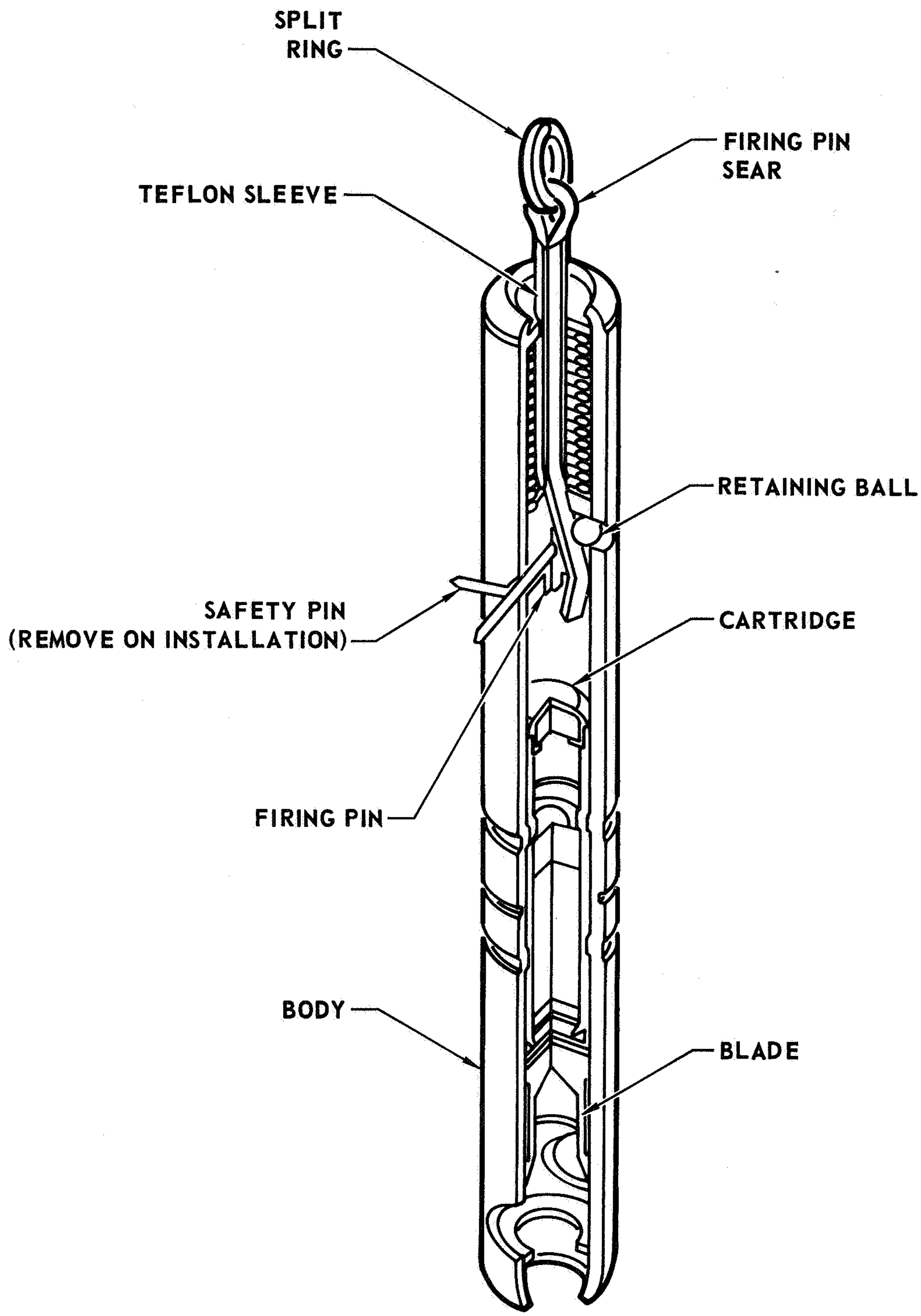


FIGURE 4.3.2.8-3

within the catapult canister. The important features of the bag are shown in Figure 4.3.2.8-4.

The deployment bag is constructed of cloth with reinforcement webbings. To insure an orderly and controlled deployment, the parachute and riser assemblies are stowed in the deployment bag in the following manner. The apex of the parachute canopy is tied with a break cord to the top inner end of the deployment bag. The canopy is folded and stowed in an accordian fashion with each folded layer being tied to restraining tabs on the interior of the bag. Four large canopy closure flaps are sewn to the inner wall of the bag and are folded down on the canopy. These flaps restrain the parachute canopy by means of a lock tab, through which the suspension line bundle is passed to form a loop lock for these flaps. This method of storage prevents the canopy from being dumped prior to the complete deployment of the suspension lines. The remainder of the suspension lines are bundled and folded in a series of loops which are stowed in a series of suspension line stowage loops sewed to the suspension line stowage flaps within the bag. The last section of the riser assembly is placed over the upper suspension line stowage flap and the bag closure flaps are sewn closed through the riser with break cord. The bag has a main riser stowage channel with two flaps which enclose and protect the riser from friction burns while the bag is leaving the catapult compartment.

Catapult Assembly - The catapult assembly is utilized to stow the parachute assembly and to provide the exit velocity required to properly deploy the parachute. The parachute deployment bag and the upper section of the risen assembly is stowed within the catapult compartment and constrained by the catapult cover. The cover is made of aluminum alloy and covered with insulation to protect the parachute from heating during descent from orbit. The cover is retained by screws located in open-ended slots. When the catapult fires, the pressure of the parachute bag against the cover causes it to "oil can" off the retaining screws, thus freeing the parachute to deploy (see Figure 4.3.2.8-5).

Nominal Performance - For the design concept, the primary aeroshell and decelerator parameters are defined in Table 4.3.2.8-1. The deployment and

PARACHUTE DEPLOYMENT BAG

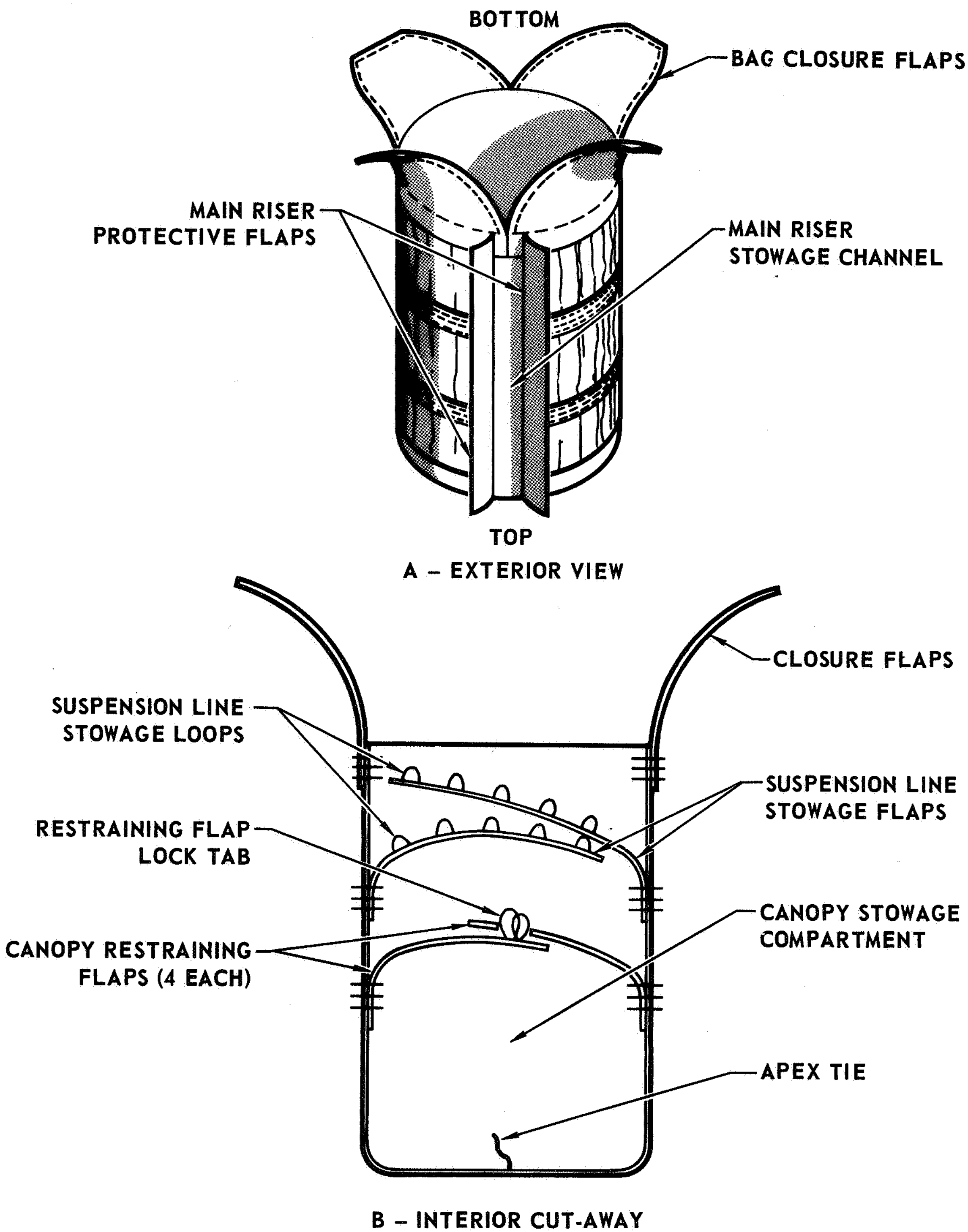


FIGURE 4.3.2.8–4

CATAPULT ASSEMBLY

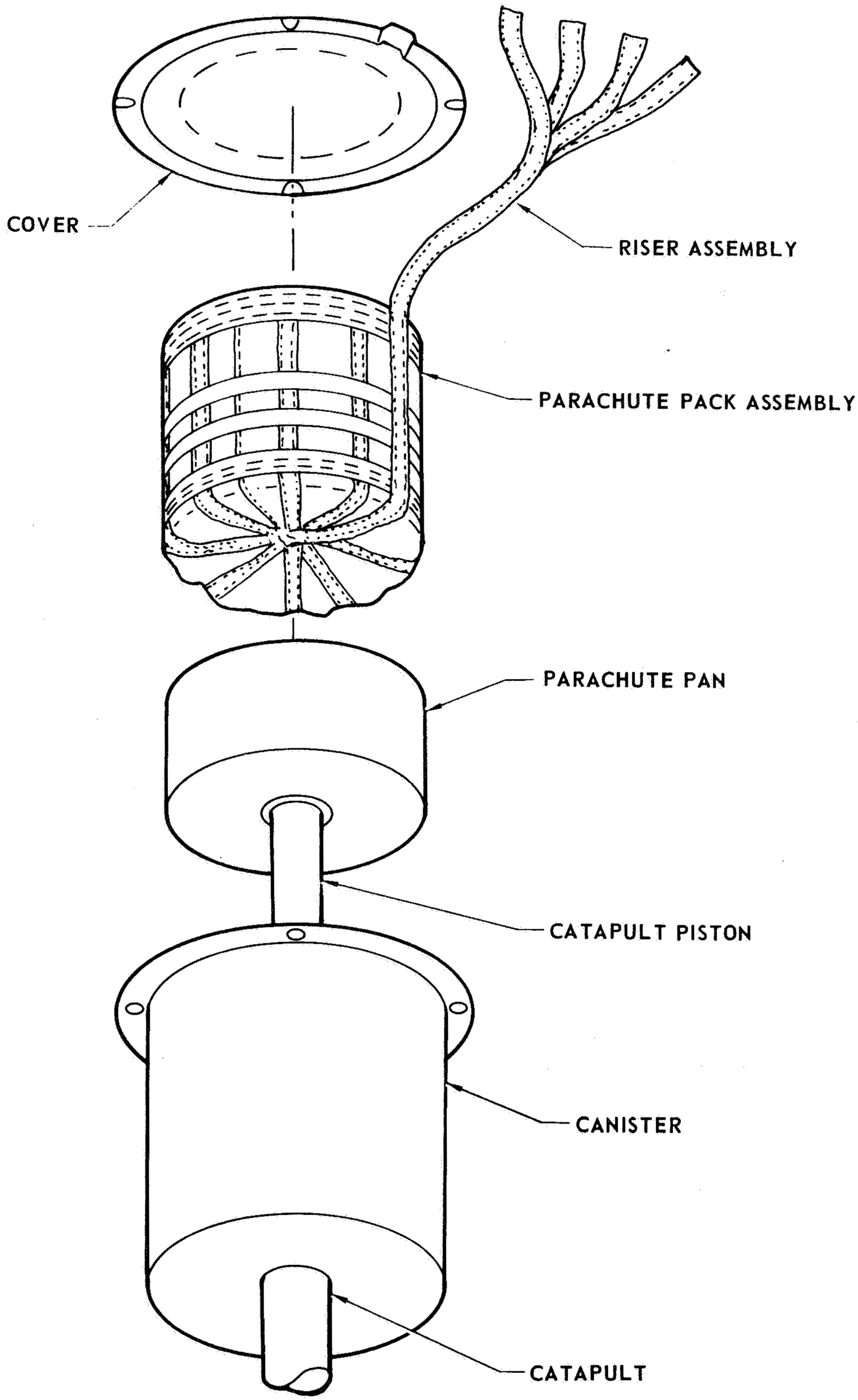


FIGURE 4.3.2.8–5

TABLE 4.3.2.8-1

AEROSHELL AND DECELERATOR PARAMETERS/CHARACTERISTICS

CONCEPT III

	VALUE
PARAMETERS	
ENTRY BALLISTIC PARAMETER $(m/C_DA)_e$	.254 $SLUGS/FT^2$
AEROSHELL DIAMETER (D_A)	11.45 FT
AEROSHELL REFERENCE AREA (A)	103.0 FT^2
AEROSHELL MASS (m)	8.70 SLUGS
AEROSHELL BALLISTIC PARAMETER @ M = 2 $(m/C_DA)_{M=2}$ (1)	.0559 $SLUGS/FT^2$
CANOPY DIAMETER (D_o)	34.0 FT^2
CANOPY REFERENCE AREA (S_o)	908 FT^2
LANDER PLUS CHUTE MASS (m_{L+C})	30.1 SLUGS
NOMINAL CHARACTERISTICS (2)	
MACH NO. AT DEPLOYMENT (M_D)	1.04
DYNAMIC PRESSURE AT DEPLOYMENT (q_D)	5.33 LB/FT^2
OPENING SHOCK LOAD	3440 LB
FILL TIME (3)	1.32 SEC
MACH NO. AT AEROSHELL SEPARATION (4)	.74
DYNAMIC PRESSURE AT SEPARATION	3.0 LB/FT^2
AEROSHELL DRAG COEFFICIENT AT SEPARATION (5)	.80
AEROSHELL BALLISTIC PARAMETER AT SEPARATION	.1056 $SLUGS/FT^2$
LANDER & CHUTE BALLISTIC PARAMETER AT SEPARATION (6)	.0474 $SLUGS/FT^2$
RELATIVE ACCELERATION AT SEPARATION	34.1 FT/SEC^2
LANDER ALTITUDE AT AEROSHELL IMPACT	5043 FT
TIME FROM DEPLOYMENT TO AEROSHELL IMPACT	44.8 SEC
LANDER VERTICAL VELOCITY AT 6500 FT ALTITUDE	306 FT/SEC

NOTES:

(1) C_D OF AEROSHELL = 1.51 @ M = 2

(2) NOMINAL DEPLOYMENT AND SEPARATION CHARACTERISTICS ARE FOR V_e = 15 110 FT/SEC, $\gamma_e = 16°$, h_D = 23,000 FT, AND VM-7 ATMOSPHERE.

(3) FILL TIME PARAMETER: $(t_F V_D)/D_o = 40$, FROM MORTAR FIRE.

(4) AEROSHELL SEPARATION OCCURS 4 SECONDS AFTER FULL INFLATION.

(5) FIGURE 3.1.3-1

(6) C_{D_o} OF CHUTE = 0.7 (BASED ON ESSENTIAL CLOTH AREA)

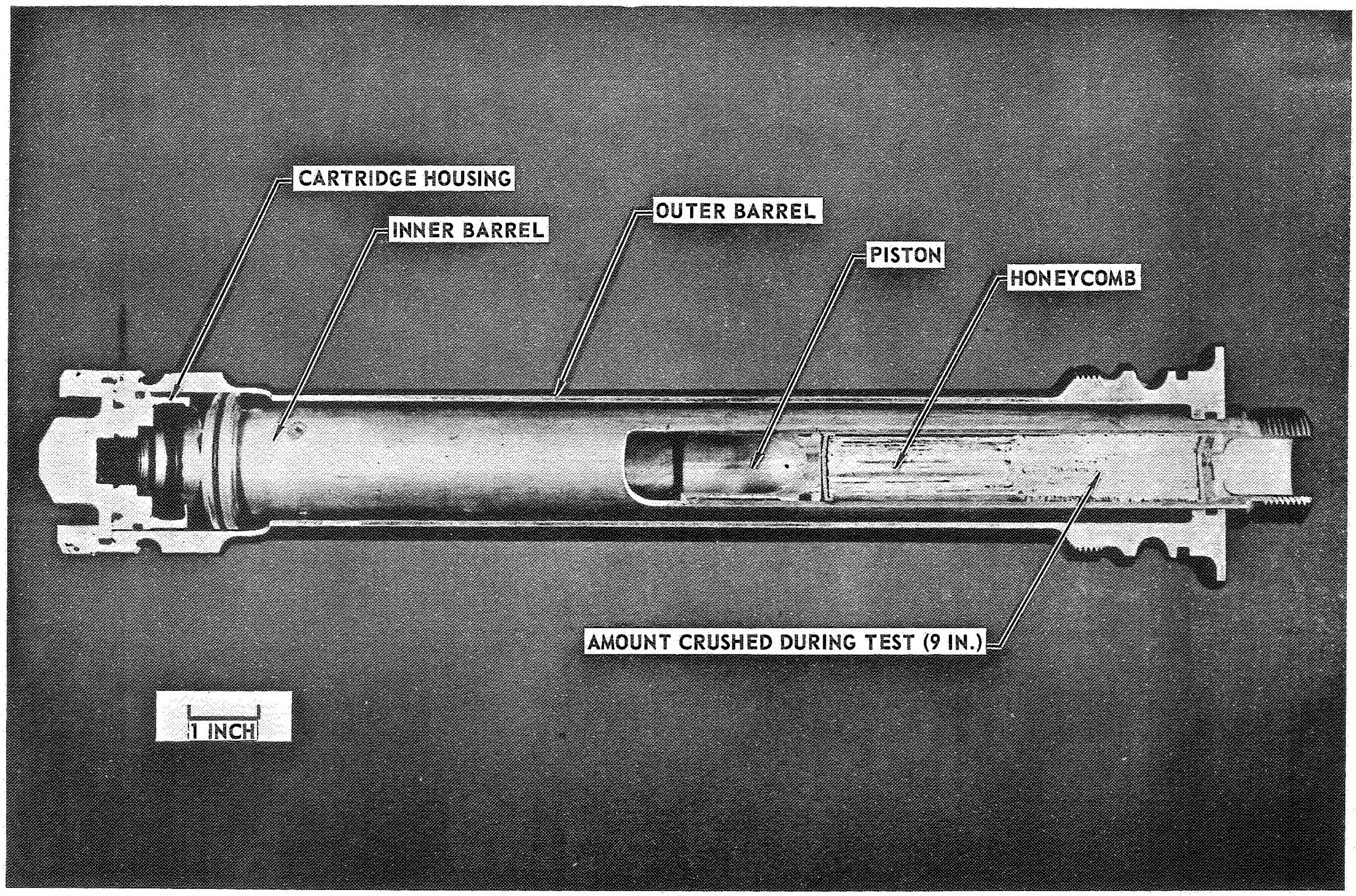
separation characteristics are also defined in the table for a nominal trajectory. The parachute design was based on Mach 2 deployment for the steep, slow entry condition of the entry corridor. Deployment of the modified ringsail parachute is initiated by radar altimeter signal at 23 000 feet. Four seconds after full inflation, the aeroshell is released and allowed to fall away from the lander. The terminal descent engines are ignited on the basis of a signal from the radar altimeter at 6500 feet; immediately after ignition, the parachute is released.

Significant Requirements and Development Status - Several interrelated trade studies have identified the advantages of using a deployable aerodynamic decelerator to increase the probability of mission success. The NASA Planetary Entry Parachute Program (PEPP) has demonstrated the operational feasibility of large parachutes up to approximately Mach 2.

The catapult design illustrated in Figure 4.3.2.8-6 has been developed and fully qualified by McDonnell Douglas for the F-111 Pod Parachute Ejection System. This catapult design is capable of ejecting weights of 120 pounds from velocities of 35 ft/sec to 55 ft/sec. The design includes a crushable honeycomb feature for precise control of chamber pressure which results in uniform acceleration and velocity control. A similar system, without the honeycomb insert, was qualified to eject 120 pounds at a velocity of 100 ft/sec.

Investigations conducted on the ability of 330 Nylon parachute material to withstand the sterilization cycle have indicated that no significant degradation in material properties result from exposure to sterilization temperature (135°C) for periods of time up to 200 hours. It is believed at this time that 330 Nylon can be considered as a worthwhile candidate material.

F-111 PARACHUTE CATAPULT CROSS SECTION



NOTE: CATAPULT SHOWN AFTER FIRING, EXCEPT INNER BARREL RETRACTED FOR PHOTO

FIGURE 4.3.2.8-6

4.3.2.9 Aeroshell. - The aeroshell, composed of structure and heat shield, provides the velocity reduction and thermal protection to the remainder of the capsule during ballistic entry into the Martian atmosphere. The aeroshell configuration, shown in Figure 4.3.2.9-1, consists of an 11.48 ft base diameter, 60 degree half-angle sphere-cone structural shell with a spherical nose radius of 2.87 ft. The heat shield on the conical section is an ablative type foamed reinforced methyl phenyl silicone, which is hard bonded to the structure. A nonablative material is used for heat shielding the spherical nose assembly. An instrument head is located at the apex of the nose section. A thermal curtain is attached to the aeroshell base area to provide thermal protection to the capsule.

4.3.2.9.1 Structure: The conical structure consists of a titanium single-faced, longitudinally corrugated shell with closed triangular aluminum rings. A fiberglass sandwich structure, used for the nose section, is attached to the conical shell at the cone-sphere tangency ring by a radial bolt pattern as shown in Figure 4.3.2.9-2. The structural weight is 102.2 lb.

The conical portion consists of a forward and aft section, mechanically joined at the payload ring. Each section is composed of an 0.008 inch titanium 6Al-4V, smooth outer skin, stitch welded to a 0.008 inch longitudinally corrugated inner skin. Four internal rings, including the payload ring, are mechanically joined to the inside of the corrugated inner skin. The rings are aluminum triangular torque boxes, composed of three caps and two beaded webs, with the shell providing the third web.

This configuration is designed without the need for a hoop load path in the skin. Each corrugation functions as an individual longitudinal beam to distribute loads normal to the shell surface into the triangular rings. The triangular rings provide stability for collapsing pressure and redistribute the lateral loads from unsymmetrical pressure into the overall shell.

The structural nose assembly, shown in Figure 4.3.2.9-2, is a reinforced plastic sandwich consisting of heat resistant phenolic (HRP) honeycomb core and phenolic fiberglass face sheets. It is assembled with a modified epoxy film adhesive (HT-435). The sandwich structure has a core which is 0.26 inch thick and face sheets that are each 0.020 inch thick. It is designed to be

AEROSHELL STRUCTURAL ARRANGEMENT

CONCEPT III

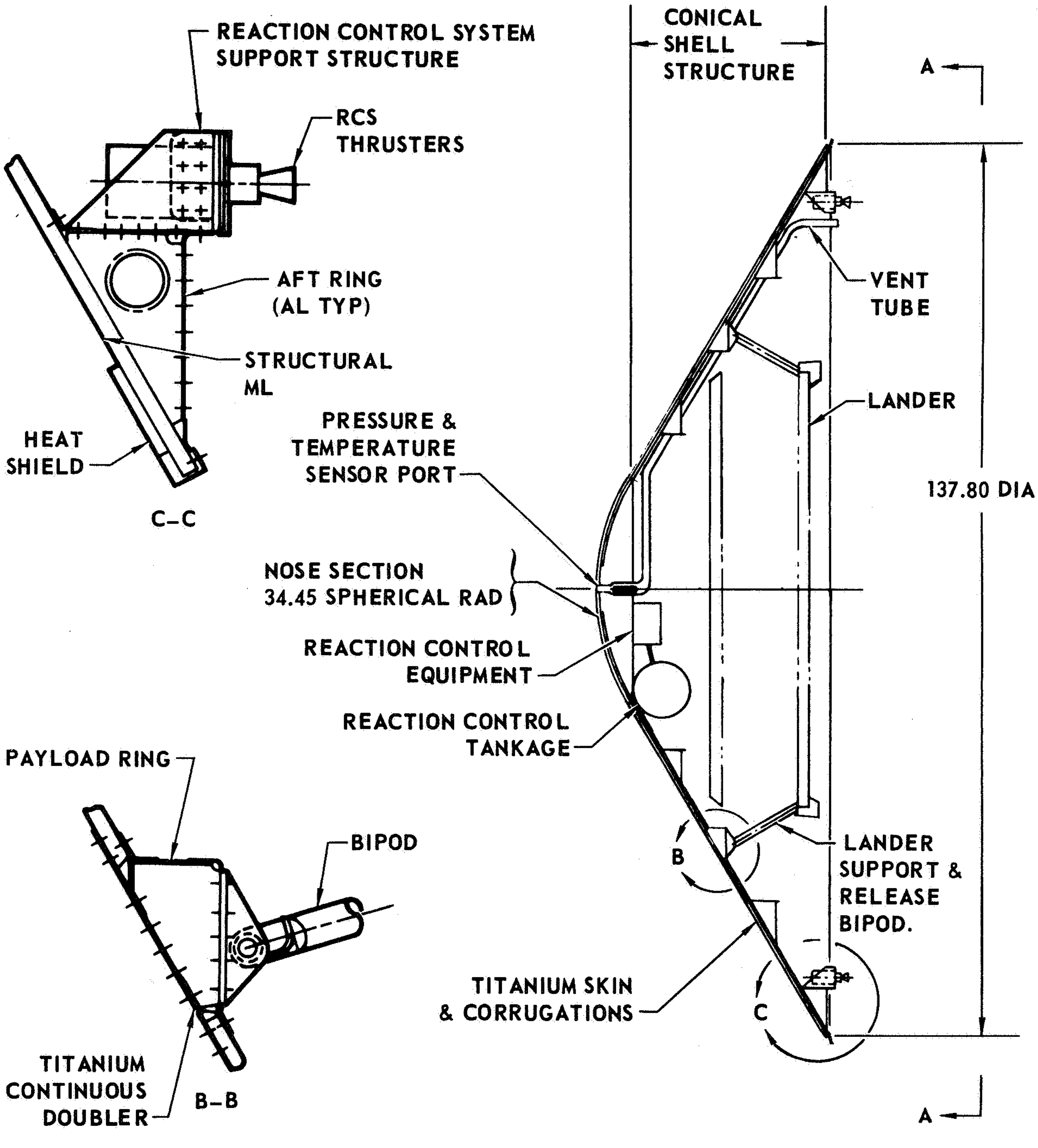


FIGURE 4.3.2.9–1

AEROSHELL STRUCTURAL ARRANGEMENT (Continued)

CONCEPT III

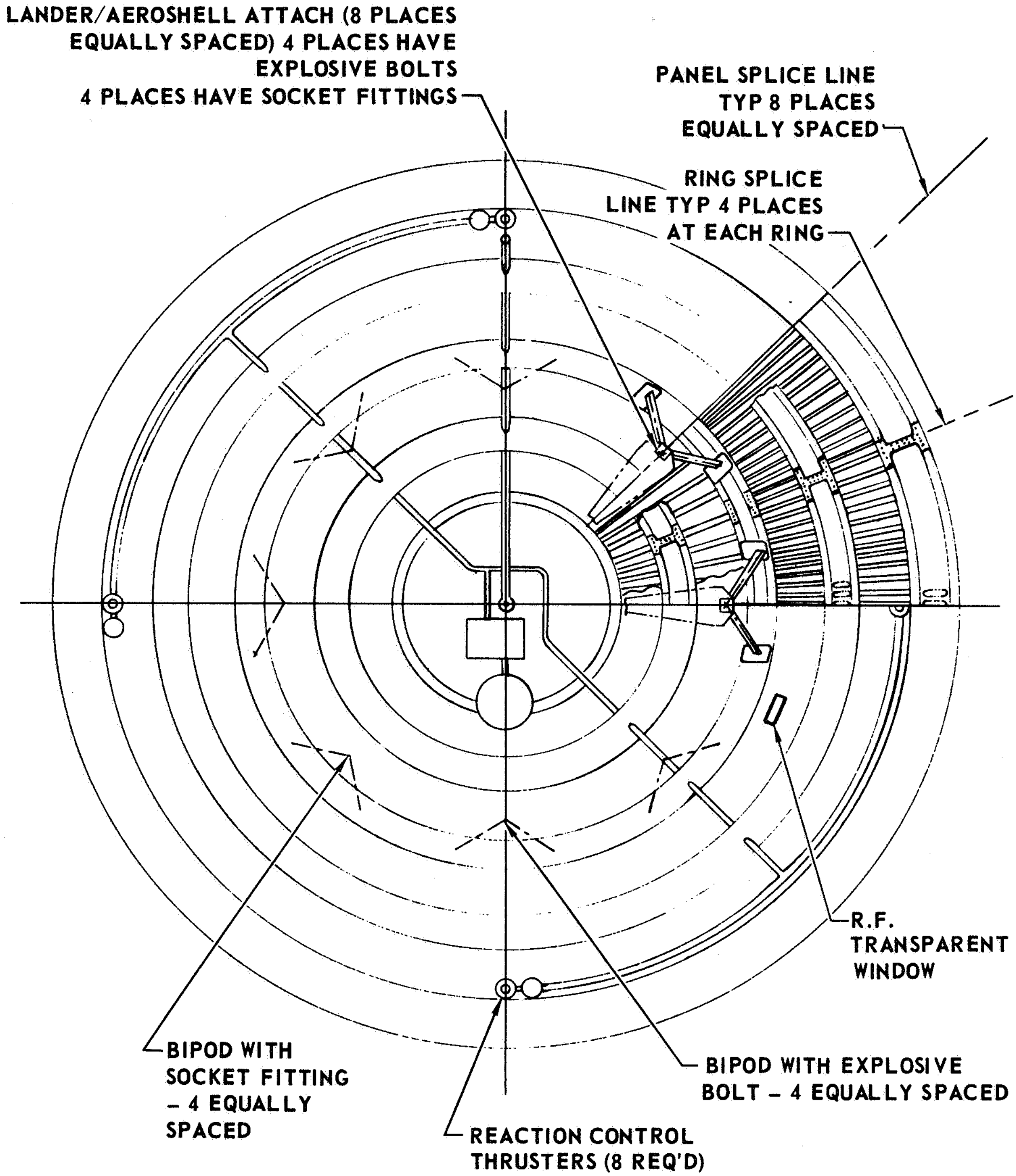


FIGURE 4.3.2.9–1

NOSE SECTION ASSEMBLY

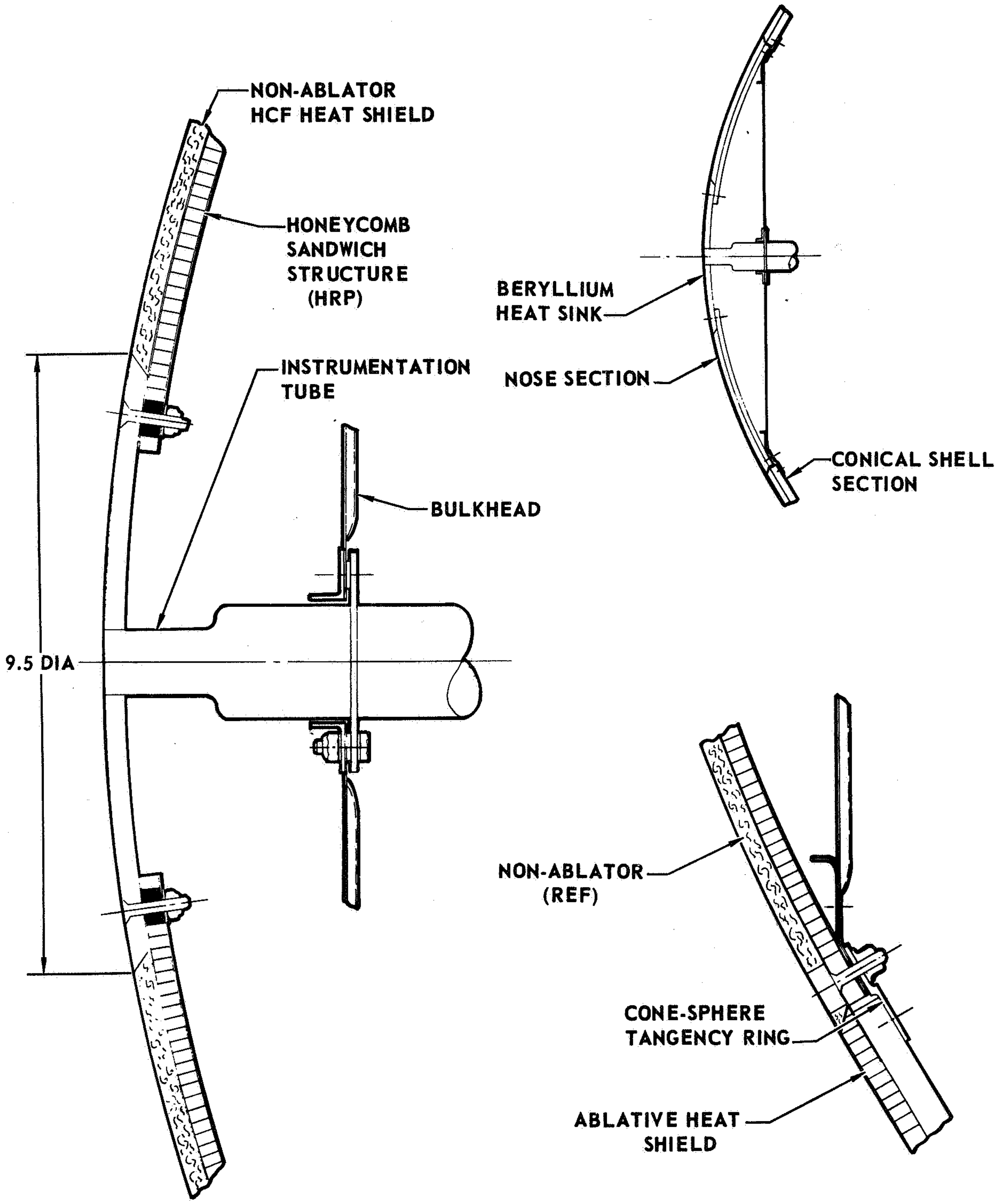


FIGURE 4.3.2.9-2

laid up, cured, and bonded directly to the inside surface of the prefabricated heat shield, in one assembly operation. Strength and stiffness is provided at the nose cap/conical shell joint by a channel section ring attached to a beaded circular disc with radial stiffening angles.

Fittings and brackets at the aft ring provide support for the eight reaction control jets and a truss attached to the cone-sphere tangency ring, provides reaction control system equipment support.

An RF transparent window, approximately 2 by 7 inches, is provided in the conical structure for the radar altimeter.

The lander is mounted to the aeroshell by eight aluminum bipod trusses located circumferentially at equal spaces along the payload ring, as shown in Figure 4.3.2.9-1. Four bipods are attached to the lander by explosive bolts and take tension, compression, and shear loads. The other four bipods mate with the lander through preloaded sockets and take only compression and shear loads. Sixteen torsion springs near the payload ring pivot all eight bipods clear of the lander when lander/aeroshell release occurs.

4.3.2.9.2 Heat shield description: The heat protection system will protect the aeroshell structure, lander, and capsule equipment during atmospheric deceleration and entry into the Martian atmosphere. The major system components are:

a) The spherical nose cap is covered with hardened compacted fibers (HCF) of aluminosilicate (density = 20 to 25 lb/ft^3). This material is non-ablative and will not contaminate the entry science ports located in a beryllium plug at the center of the nose cap surface.

b) The conical aeroshell surface is covered with a low density foamed silicone ablator, MDC S-20T (density = 18 to 20 lb/ft^3)

c) The HCF and S-20T are attached to the aeroshell structure with HT-424 high temperature adhesive using techniques similar to those developed for the Gemini heat shields.

d) The aft regions of the aeroshell and lander are protected with a thermal curtain made with silica cloth.

o Design Requirements and Constraints - The heat shield and curtain, after being subjected to sterilization/decontamination, ground handling, shipping, ascent, trajectory adjustment, de-orbit loads, and long-term temperature and vacuum exposure, will provide heat protection for Mars entry.

o Non-Ablative Nose Cap - The design thickness of the nose cap limits the backface temperature to 640°F. The maximum backface temperature acceptable for structural design and bond compatibility is 735°F. Thus, the nose cap provides a thermal design margin of 95°F. The nose cap must not emit any gaseous products, during entry heating, that will compromise the mass spectrometer data on atmospheric samples.

o Ablative Heat Protection - The heat shield thickness for ablative heat protection has been sized to be compatible with the aeroshell structural temperature limits. The design heat shield thickness limits the backface temperature to 640°F. The maximum backface temperature for structural design and bond compatibility is 800°F, thus allowing a thermal design margin of 160°F.

o Thermal Curtain - The thermal curtain will enclose the base of the capsule to protect the surface lander, aeroshell structure and equipment from wake heating during entry, and heating during retro-rocket firing. It will help to minimize heat loss during orbital descent. The curtain must have short-term (3 minute), high temperature capability to 1500°F, and be designed so that it will not interfere with separation of the lander portion of the flight spacecraft.

Design Entry Conditions for 1973 Flight - The entry conditions are shown in Table 4-1. Angle of attack below 800 000 feet is restricted between 0° and 30° to avoid contamination of entry science measurements.

o Physical Characteristics - The heat shield consists of the non-ablative nose cap and the ablative heat protection material. The heat shield total weight is 90.9 lb. The thermal curtain weighs 14.6 lb.

o Non Ablative Nose Cap - The nose cap is a spherical segment with a radius of 2.87 feet (34.4 inches) and a chord diameter of 2.87 feet (34.4 inches). The non-ablative material is a hardened compact fiber composite consisting of aluminosilicate fibers with a collodial silica binder that is bonded to the structure with HT-424 adhesive. An instrument head imposes the use of a heat sink on the nominal stagnation point. The heat sink is a 9-1/2 inch diameter disc machined from beryllium.

o Ablative Heat Protection - The ablative heat protection material on the conical surface is the McDonnell S-20T low density silicone ablator foamed into prebonded fiberglass honeycomb that is bonded to structure with HT-424 adhesive.

o Thermal Curtain - The thermal curtain will be silica cloth with a black coating on both surfaces.

Operational Description - The heat shield and thermal curtain are passive devices, and as such, the Operational Description will be limited to a relation of function to mission sequence.

Following separation of the entry portion of the flight spacecraft and the deorbit impulse, the heat shield functions as an aerodynamic decelerator and to provide thermal protection from the resulting net heat pulse. The thermal curtain protects the capsule from base heating during deorbit motor firing and wake heating during entry.

The aeroshell, including heat shield, is jettisoned after parachute deployment, while a part of the thermal curtain remains over the lander.

The thermal curtain is stripped away from the lander along with the parachute as the lander continues descent.

Performance Objectives - The heat shield will limit the backface temperature to 640°F. It shall also minimize interaction with the other flight capsule subsystems, due to out-gassing prior to entry and products of decomposition during entry. The thermal curtain will isolate the aeroshell structure, surface lander, and aeroshell equipment from the effects of base heating.

Development Status - All materials selected for the entry heat protection system are available, processing procedures are defined, and fabrication methods have been developed in conjunction with the selected substructures. Additional scale-up development is required for the HCF nose cap material to demonstrate that fabrication methods are satisfactory for the larger sized components. Additional tests of thermodynamic, mechanical, and entry heating characteristics shall be measured to form a complete data base for designing the heat shield.

4.3.2.10 Landing system. The McDonnell Uni-Disc lander configuration is a large flat disc having a low landing silhouette with a low c.g. and high landing stability. This shape evolved primarily from the requirement for stable landing on extreme surface slopes and contours, and the desire to standardize the concept for varied missions. This system weighs 138.3 lb.

The lander consists of a structural base platform to which are mounted the surface payload, the terminal descent propulsion subsystem, and other Capsule support subsystems required to control descent and landing. An impact attenuation subsystem is attached to the underside of the base platform. The general arrangement is shown in Figure 4.3.2.10-1.

This configuration has been designed to the following constraints:

a) Provide stability, strength and ample footprint to make successful landings under the following conditions:
   - o Slopes to 34°
   - o Ridges with slopes $\pm 34°$ to horizontal
   - o Cones with slopes $\pm$ 34° to horizontal
   - o Surface bearing capacities of 6 psi to infinity
   - o Rocks of 5 in. diameter or smaller
   - o Vertical and horizontal velocities of 20 ft/sec and 5 ft/sec, or 16 ft/sec and 10 ft/sec respectively
   - o Constrain the impact felt by the surface payload to $<$ 25 g

b) Provide a low overall c.g. to improve stability on landing.

c) Fit inside the 11.84 foot diameter aeroshell.

d) Fit sufficiently forward in the Aeroshell to maintain and aerodynamically stable c.g. for the entry vehicle.

e) Provide a stable platform after landing.

f) Provide good structural ties and load distributions between the Aeroshell, the lander and the adapter structure.

g) Provide an unobstructed landing radar antenna location, preferably

CONCEPT III
LANDER GENERAL ARRANGEMENT

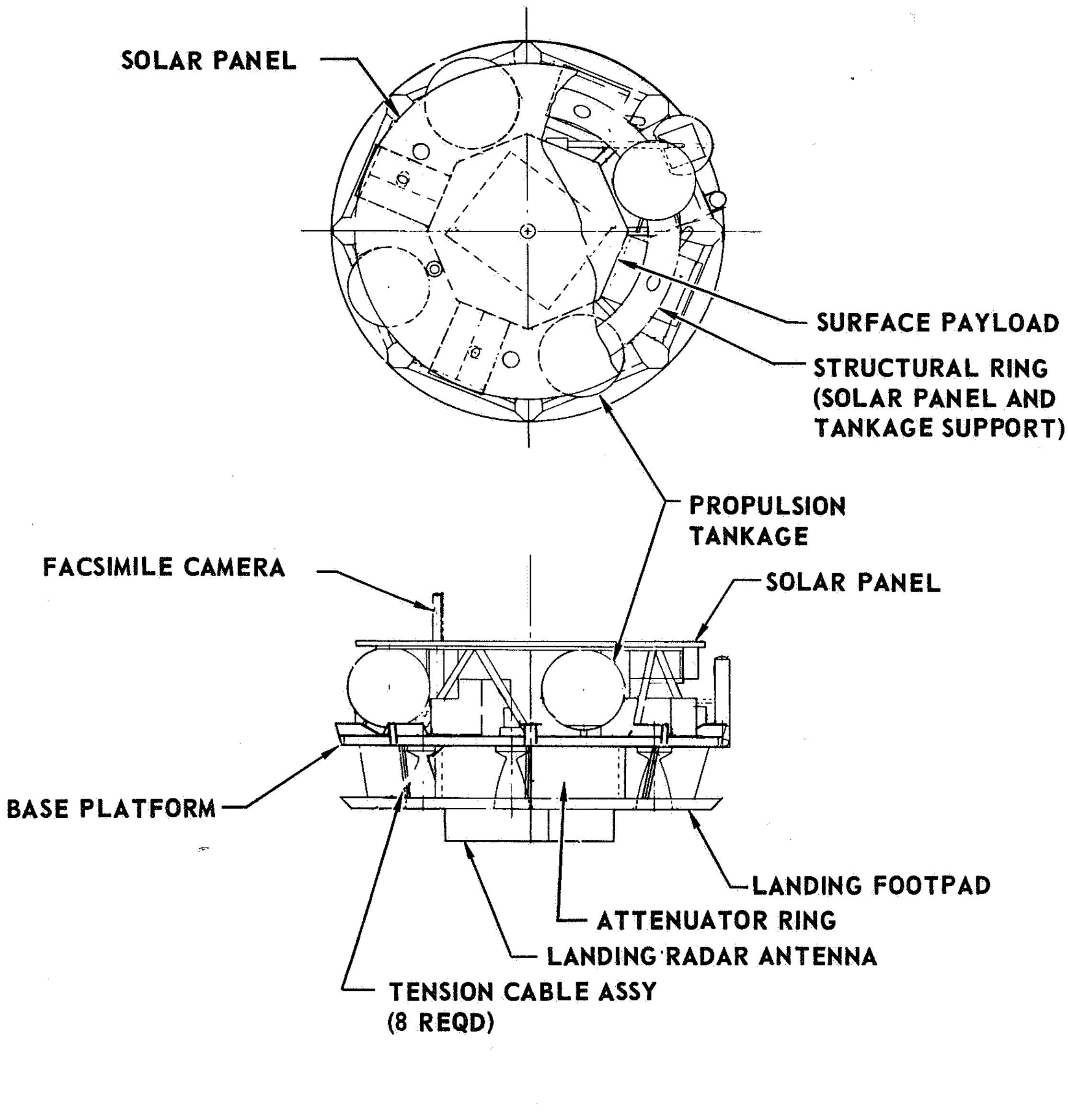


FIGURE 4.3.2.10-1

bottom center.

h) Provide a descent propulsion engine installation with ample moment arms, located so as not to impinge on other parts of the lander.

i) Be compatible with clean separation from Aeroshell.

j) Provide a simple, passive system that is easily sterilized and can function after long time exposure to a hard vacuum.

k) Use materials and techniques that are within the state of the art to minimize development risk and cost.

l) Standardize on as many subsystems as is practical.

m) Provide versatility and growth capabilities.

4.3.2.10.1 Physical characteristics: The landing footpad is 66.3 inches in diameter. This pad is 2.0 inches thick with a turned-up outer lip to facilitate sliding over small obstacles. The footpad is made up of titanium radial beams, rings, and lower skin. Sufficient structural rigidity is required to uniformly distribute impact loads to the crushable attenuator when the lander lands on one edge or on peaks and ridges. The nozzles of the terminal descent engines thrust downward through three 9.0 inch diameter holes between the radial beams of the footpad.

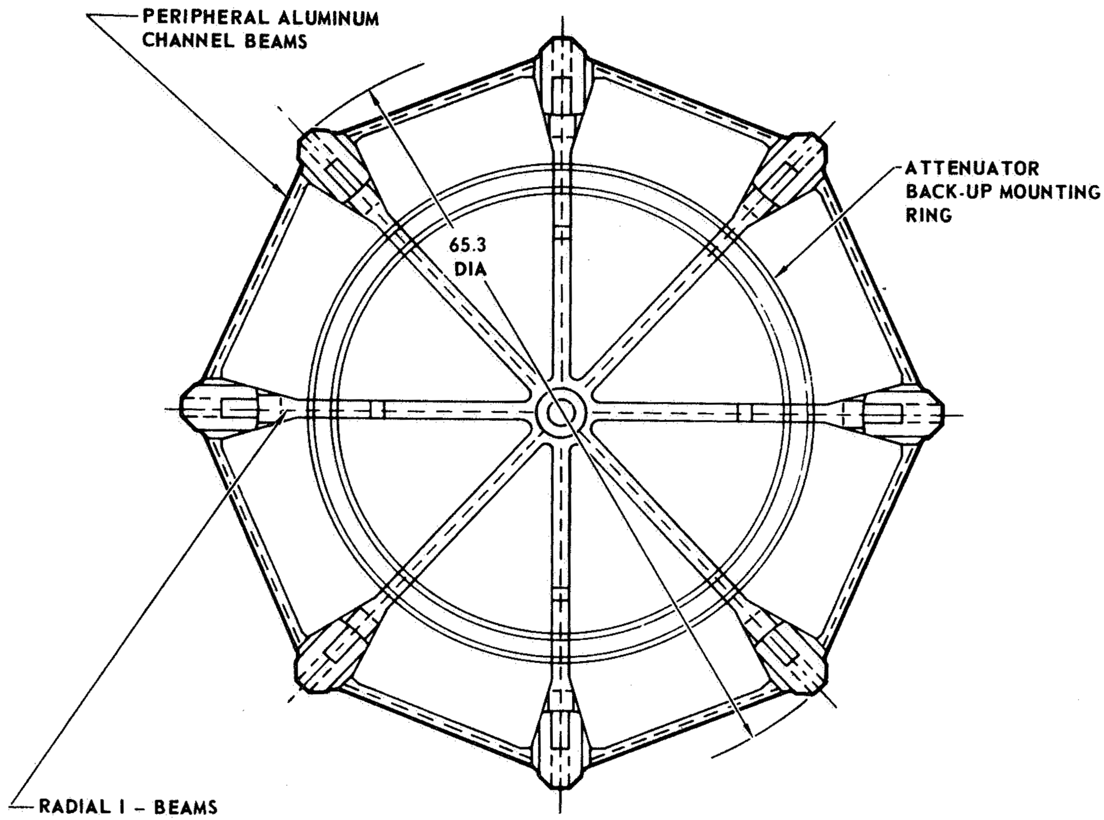
The base platform is made up or eight titanium I-beams, 2 inches deep, with 1.3 inch caps, equally spaced in a radial spoke arrangement as shown in Figure 4.3.2.10-2. Maximum diameter is 65.3 inches.

The impact attenuator is installed between the landing footpad and the base platform. It is a crushable cylinder of aluminum honeycomb, such as "Trussgrid", 10 inches high and 45.4 inches in diameter with 1.6 inch wall thickness. This attenuator is bonded and mechanically keyed to both the footpad and the base platform through structural channel rings to insure the transfer of landing loads. The attenuator material has crushing strength of 75 psi and a density of 3.3 lb/cu ft.

Eight tension cable pulley assemblies are mounted to the ends of the radial I-beams of the base platform outboard of the attenuator. These cables tie

LANDER STRUCTURAL ARRANGEMENT

CONCEPT III



BASE PLATFORM STRUCTURE

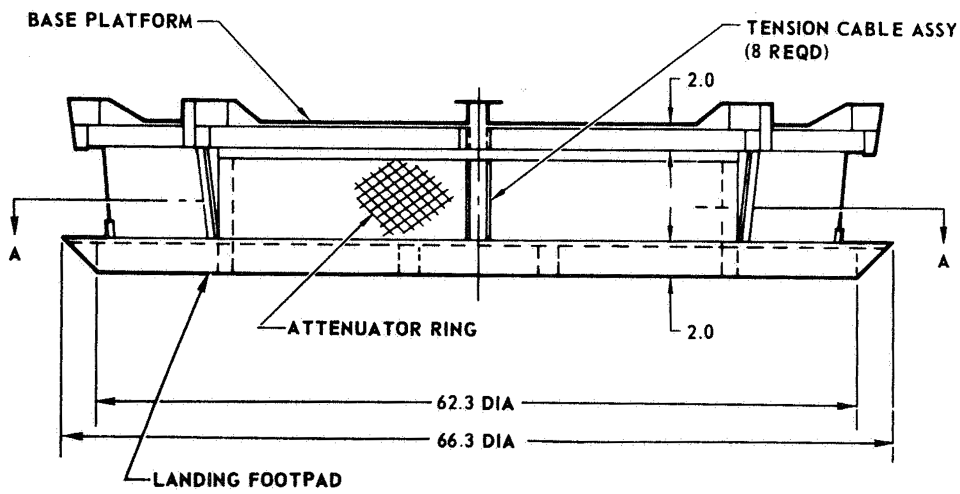
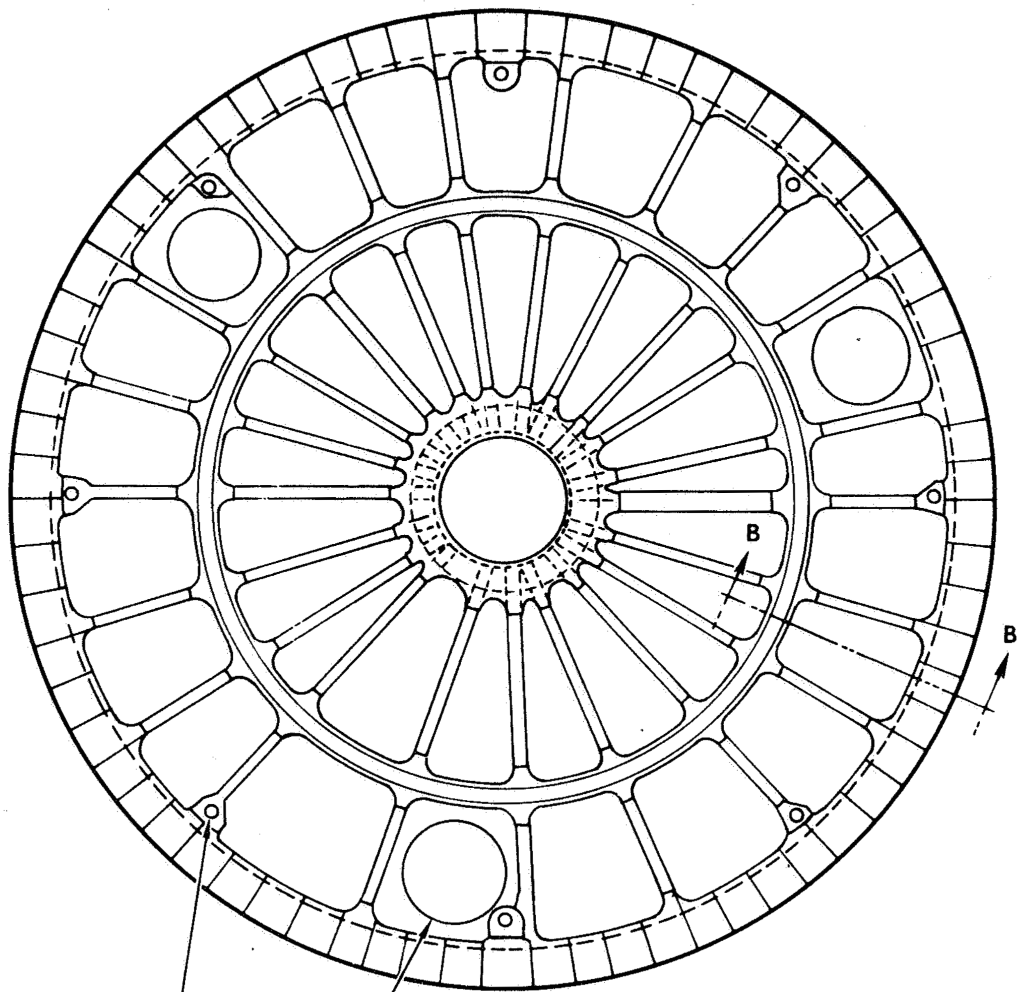
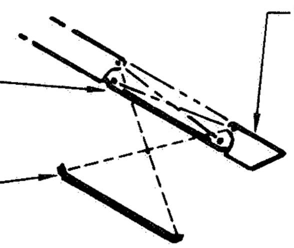


FIGURE 4.3.2.10–2

LANDER STRUCTURAL ARRANGEMENT (Continued)



VIEW A–A
FOOTPAD STRUCTURE



STABILIZING LEG GEOMETRY
VIEW B–B
(TYP 3 PLACES)

FIGURE 4.3.2.10–2

the footpad to the base platform and serve as pivot points for the rigid footpad when it lands on the opposite edge. By forcing the footpad to rotate about this point the entire crushable attenuator is put into compression, thus eliminating tension loads across the bond between the attenuator and adjoining structure. As the attenuator crushes, spring loaded cable pulley assemblies take up the cable slack and one-way ratchets prevent cable lengthening. This insures a repeat attenuation capability should the lander bounce and land on the opposite edge. In addition, this continuous snugged-up condition of the structure and the impact attenuator insures attenuation of the horizontal velocity loads through the attenuator. (These horizontal loads may also be dissipated, in part, through friction between the footpad and the Mars surface when sliding occurs.)

Secondary structures are used to support various electronic equipment as well as the terminal descent propulsion tankage.

Eight aluminum channel beams connect the ends of the eight primary radial I-beams of the base platform, forming a peripheral octagon shape. The terminal descent system tanks are supported from the radial I-beams of the base platform by tubular struts in a manner to allow for thermal variations.

The terminal descent motors are mounted to the radial I-beams of the base platform. The terminal descent propulsion system is shown in Figure 4.3.2.10-1.

4.3.2.10.2 Landing system operation: At a point 10 feet above the Mars surface the terminal descent engines are cut off allowing the lander to free fall from that height. Because of the $\pm 5°$ variation in trajectory angle and a $\pm 5°$ attitude error from uneven engine tail off, the lander may be rotated as much as 10° from horizontal on impact. This 10° tilt, when coupled with landing on a 34° slope, imposes an equipment installation clearance line of 44°, as shown on Figure 4.3.2.10-3.

In the case of a straight flat impact at zero horizontal velocity on a level surface, the footpad distributes the loads uniformly to the crushable Trussgrid. The Trussgrid attenuator absorbs sufficient energy while stroking to reduce the loads felt by the Surface payload and associated equipment during

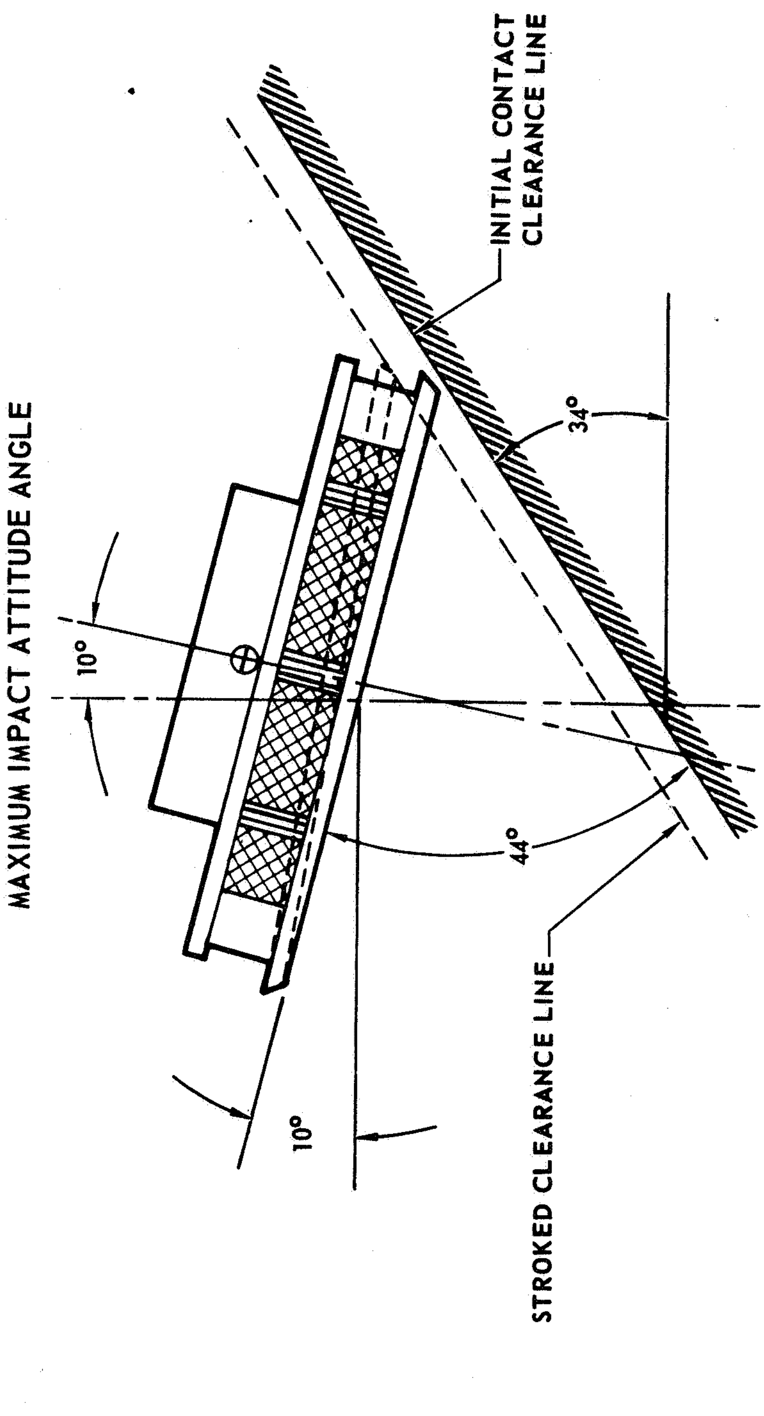


FIGURE 4.3.2.10-3

landing impact to no more than 25g. In this case the eight tension cable assemblies merely roll up the slack cable created by crushing the attenuator.

The landing radar antenna which is mounted to the bottom center of the footpad, the frangible nozzles of the terminal descent engines, and the radar altimeter antenna are crushed during landing.

For the case of an off-center landing on a slope, ridge or peak, coupled with horizontal velocity the operation is as follows:

The off-center vertical load outboard of the attenuator ring is introduced through the footpad to the attenuator. Because the footpad is a stiff structure it would begin to rotate about the attenuator nearest the impact point if not restrained by the tension cable assemblies. This rotation of the footpad would put tension loads on the bond between the attenuator and the footpad on the side opposite the impact point. This would cause footpad/attenuator separation and only local crushing of the attenuator ring. Use of the tension cables prevents this separation and forces the crushable ring to compress over its entire area.

The one-way ratcheted cable pulleys keep the footpad tight against the attenuator and keep the system operational during bouncing or sliding. The base platform provides a rigid structural backbone and reacts the loads introduced by the attenuator and the tension cables through the radial I-beams.

When the lander has come to rest, short stabilizing legs are extended to minimize disturbances during the landed operations. These legs are mounted flush with the underside of the impact footpad, so as not to be damaged on impact or during skid out. The shoe of the stabilizer is attached to a spring loaded scissors mechanism, which is held in the stowed position by a bolt extending through a pyrotechnic bolt cutter. The three stabilizers, when simultaneously. released extend downward until meeting a 50 pound resistance, or until they are fully extended to 12 inches. A one-way ratchet mechanism locks the stabilizer at any point through its stroke, allowing it to support its full portion of the lander weight. Figure 4.3.2.10-4 shows the mechanical details of the landing system.

MECHANICAL DETAILS

UNI-DISC LANDER STABILIZING LEG

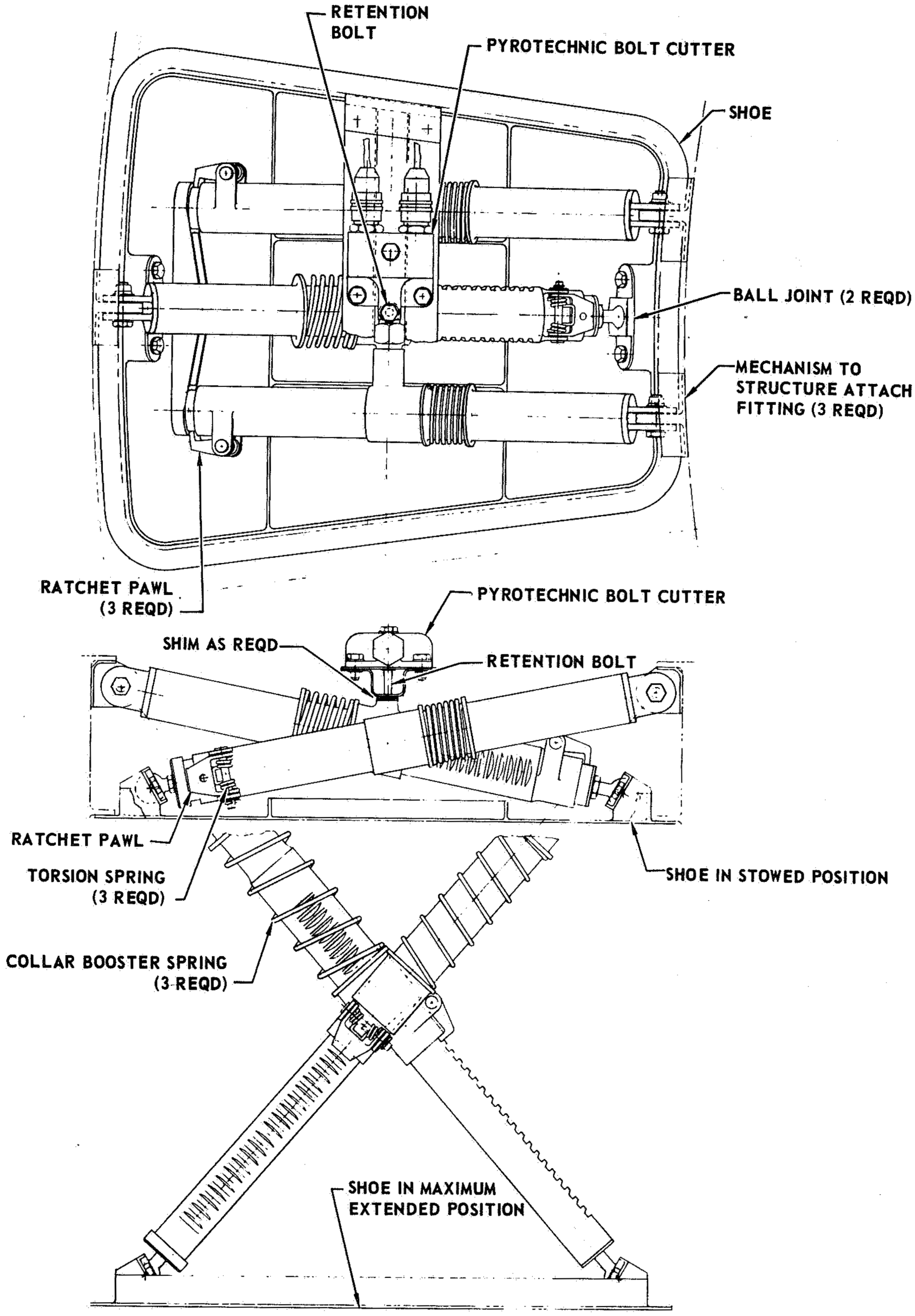


FIGURE 4.3.2.10–4

MECHANICAL DETAILS (Continued)

PULLEY-CABLE ASSEMBLY-UNI-DISC LANDER, IMPACT ATTENUATION SYSTEM

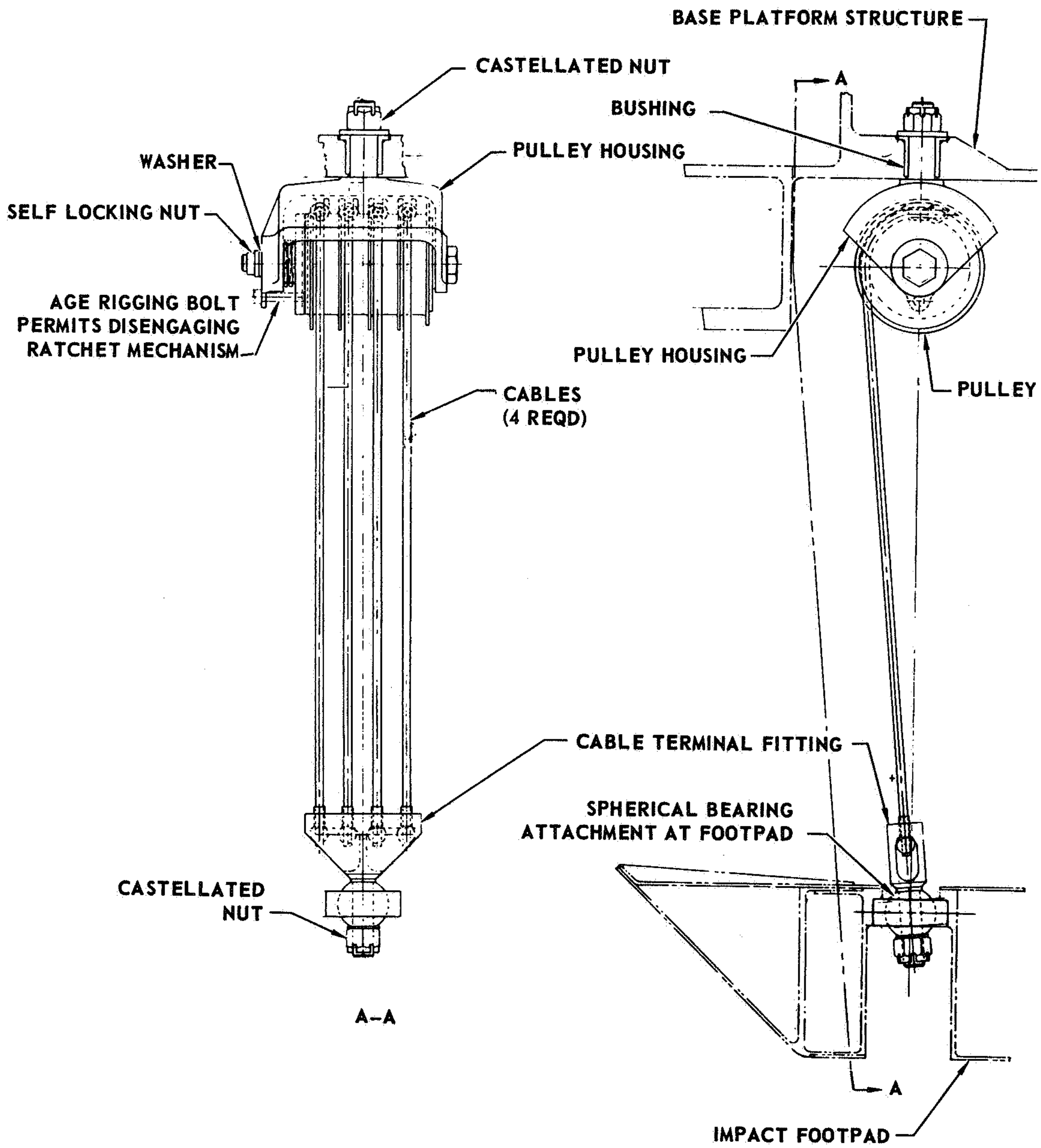


FIGURE 4.3.2.10-4

4.3.2.10.3 Interfaces: There are two separation interfaces on the lander. These are aeroshell to lander, and lander to adapter.

Aeroshell to Lander - This system consists of eight separate bipod trusses which are attached and hinged to the aeroshell at sixteen places on the second (payload attachment) ring. These bipods are spring loaded to swing outboard to provide maximum clearance for lander separation. The apexes of the bipods attach to the ends of the radial I-beams of the lander structural base. Four of the bipods, spaces 90° apart, are tied to the lander I-beams (by explosive bolts) to carry both tension and compression loads. The remaining four bipods have "ball" ends which are seated in sockets at the ends of the base platform I-beams and allow transmittal of compression loads only. All eight bipods are capable of carrying side or shear loads in the plane of the bipod. All bipods are adjustable to make up for manufacturing tolerances and meet preload requirements. These bipods are shown in Figure 4.3.2.10-5. The separation provisions weigh 14.3 lb.

Lander to Adapter - On the upper surface of the ends of the eight primary radial I-beams of the lander base platform structure are eight explosive bolts that attach the lander to the adapter truss section, as shown in Figure 4.3.2.10-6. This equipment weighs 10.3 lb.

Lander Separation - After deorbit motor firing and jettisoning, the parachute is deployed at 23 000 feet. The parachute catapult reaction loads are carried through the remaining section of the deorbit motor mounting truss structure. Parachute harness assembly lines are attached to the lander base platform at the same four points as the deorbit motor mounting truss. After the parachute is deployed, the aeroshell to lander interface ties are broken by firing four pyrotechnic bolts allowing the aeroshell and the lander to be separated by differential aerodynamic drag. At an altitude of 6500 feet, 0.5 seconds after the successful starting of the terminal descent engines, the parachute is jettisoned by activating the four explosive bolts attaching the deorbit motor support structure to the lander. The parachute carries the deorbit motor mounting truss and the thermal control curtain away with it.

AEROSHELL TO LANDER STRUCTURAL INTERFACE

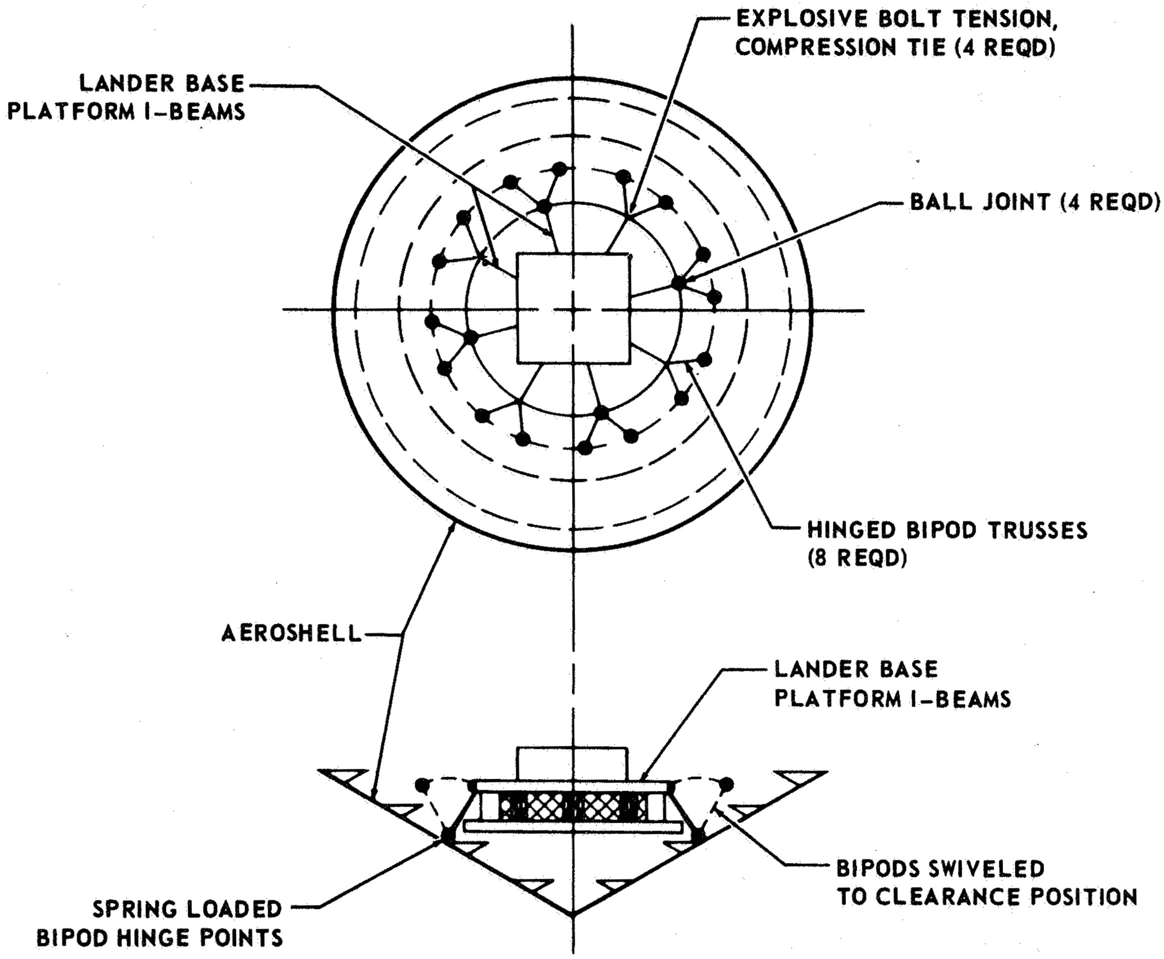


FIGURE 4.3.2.10-5

ADAPTER TO LANDER STRUCTURAL INTERFACE

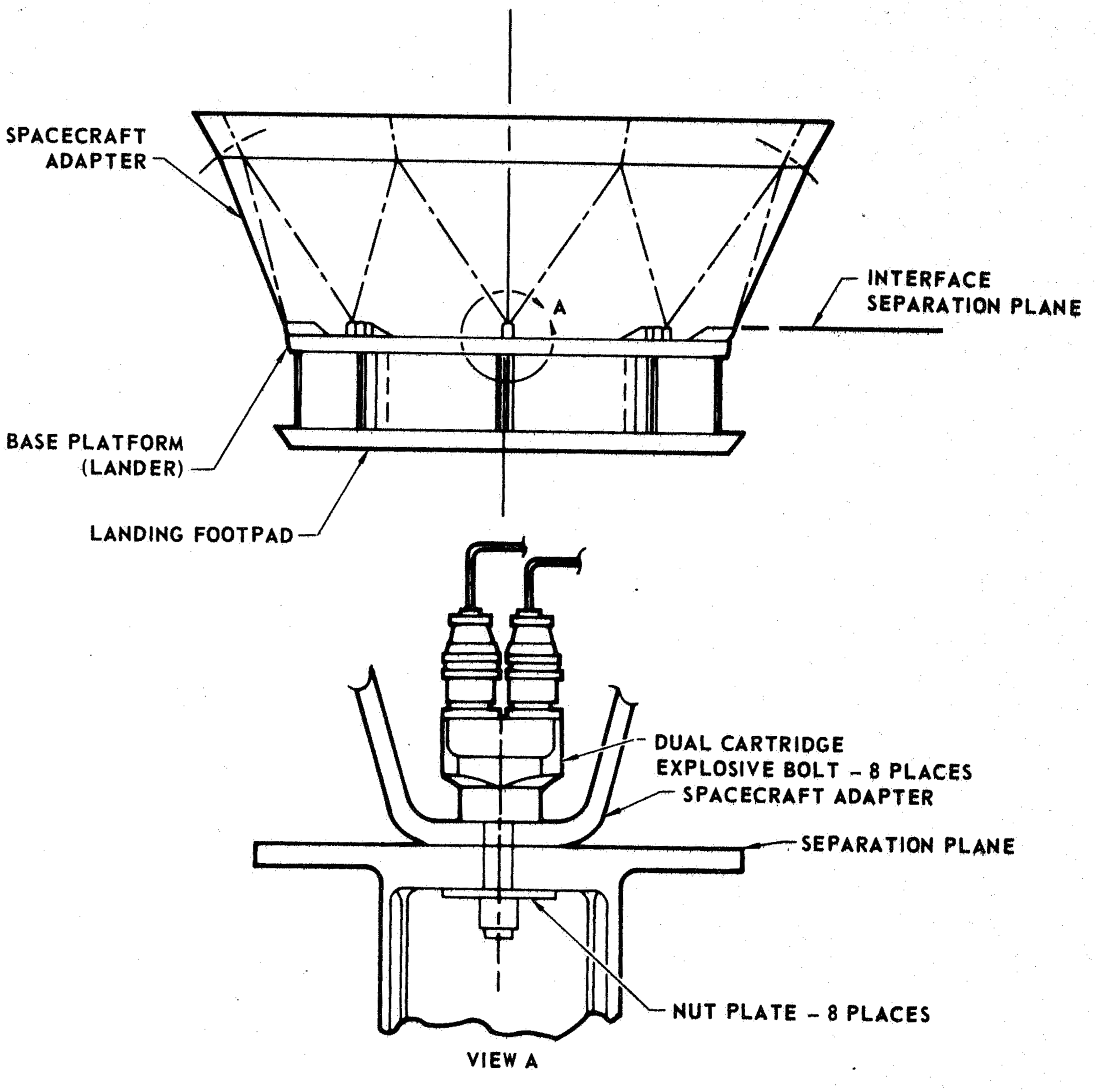


FIGURE 4.3.2.10-6

4.3.2.10.4 Impact sensor: The impact system incorporates a mechanical sensor to signal impact. This sensor is a probe or break wire subsystem that utilizes the stroking of the attenuator or the rotation of the tension cable pulley assemblies to initiate the function. This subsystem is rugged and unsusceptible to premature operation.

4.3.2.11 Canister and adapter. - The function of the sterilization canister is to maintain sterile conditions of the lander, capsule and associated equipment from terminal sterilization until canister separation. The function of the adapter is to join the capsule to the canister and to provide capability for transferring capsule inertia loads to the canister/adapter interface.

The canister configuration, shown in Figure 4.3.2.11-1 consists of forward and aft aluminum semi-monocoque shell structures mechanically joined at the field joint ring. The canister has a maximum diameter at the separation plane of 145.30 inches and a maximum height of 90.00 inches. A separation device is contained between each canister section field joint ring flange. In addition, pressurization and venting (P&V) equipment is installed in the canister to prevent excessive internal canister pressure.

The adapter configuration, shown in Figure 4.3.2.11-2, consists of a welded truss assembly mechanically attached to the lander at the adapter/capsule interface and to the canister at the canister/adapter interface.

4.3.2.11.1 Canister systems: The canister system, which weighs 140 lb, consists of structure, separation, and pressurization and venting.

Structure - The canister structure is composed of two major sections or assemblies, as shown in Figure 4.3.2.11-1. Each section consists of an 0.012 inch aluminum spherical segment shell, an 0.040 inch aluminum toroidal segment, and a field joint ring. Each field joint ring is a 7075-T6 aluminum machining. Titanium bolts are used to mechanically join the sections at the field joint ring. External aluminum rings and meridional stringers stiffen the shells and provide support for the insulation blanket covering the external surface. A fiberglass window is provided in the forward section for checkout of the capsule radar altimeter after sterilization.

The aft section mates with the adapter and the interface cone in addition to the forward canister section. Bolts through the canister wall are used to attach the adapter to fittings on the interface cone. The other end of these fittings are bolted to the orbiter. The interface cone transfers loads from the canister/adapter interface to the orbiter. Use of the

CANISTER STRUCTURAL ARRANGEMENT
CONCEPT III

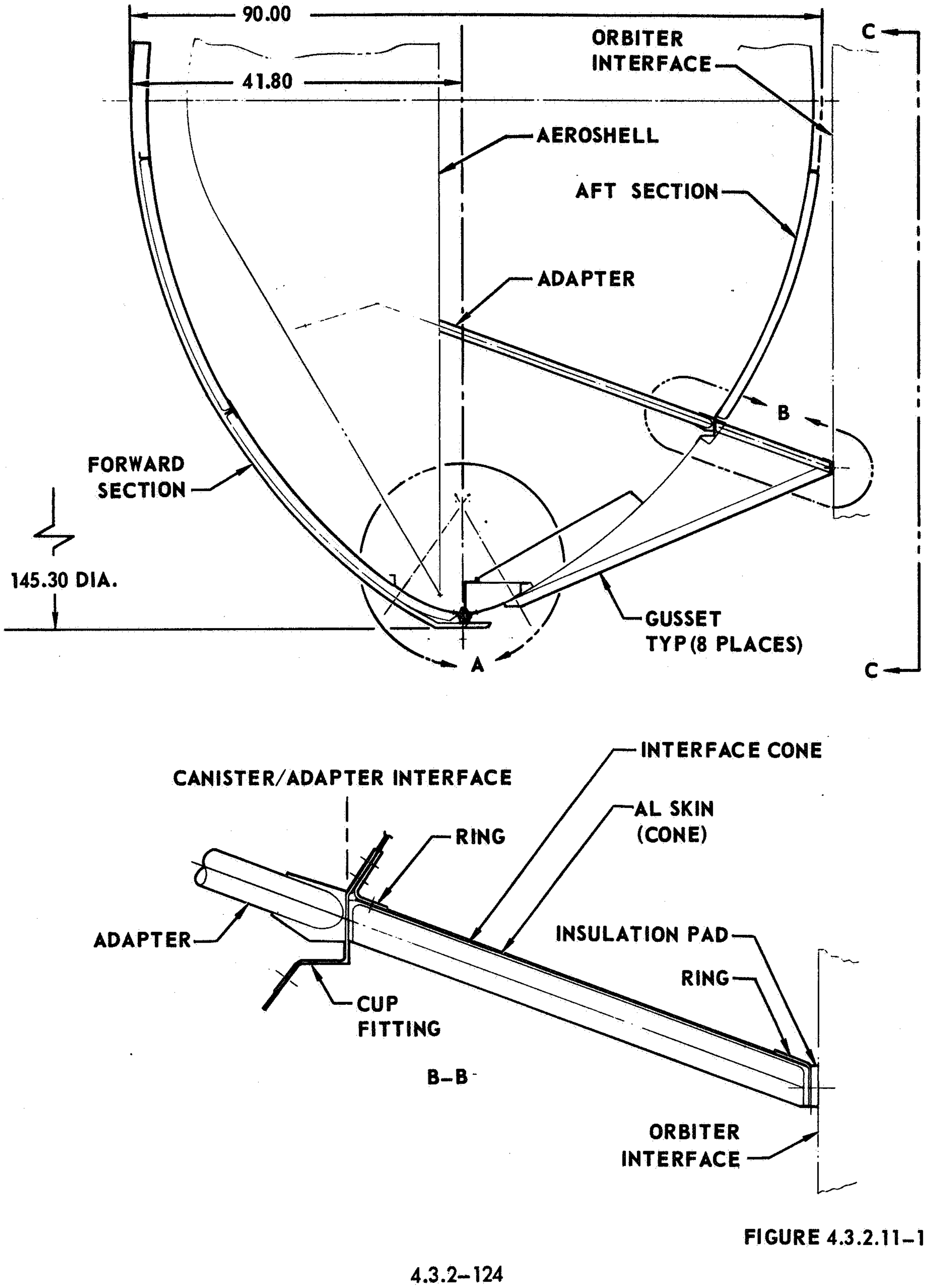
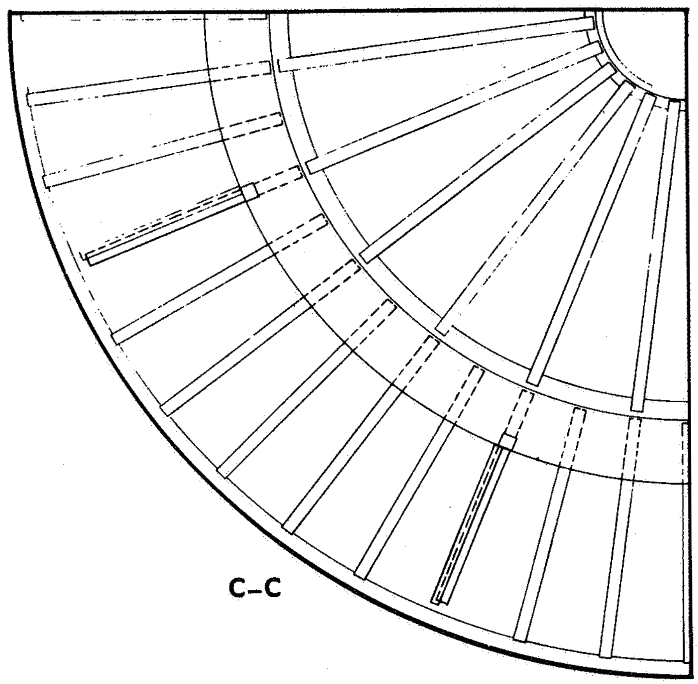
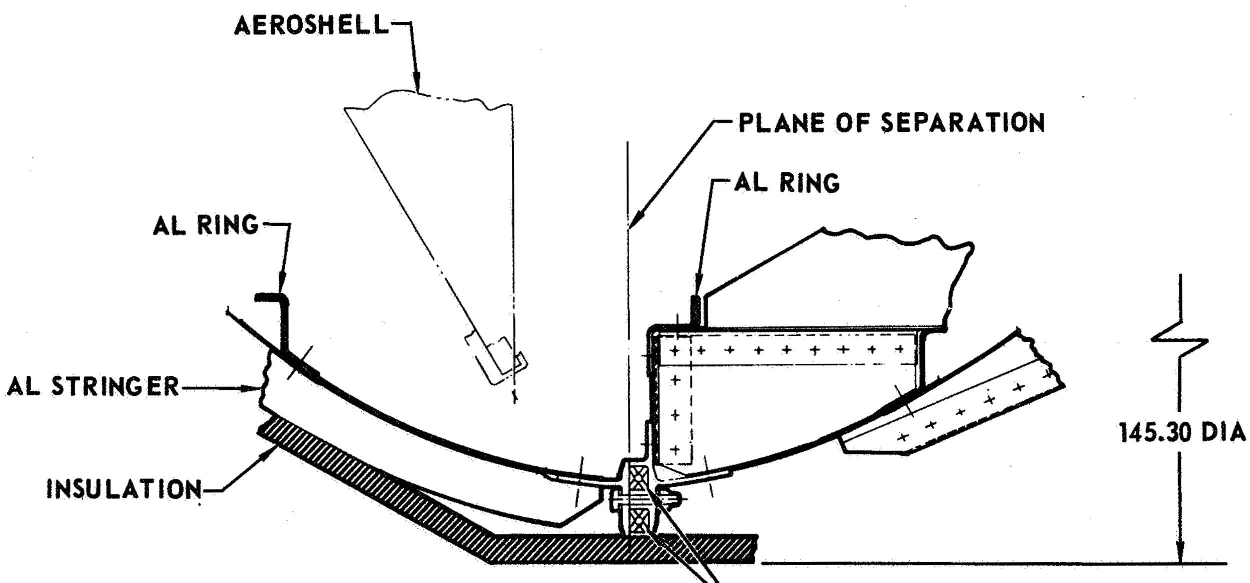


FIGURE 4.3.2.11-1

CANISTER STRUCTURAL ARRANGEMENT (Continued)
CONCEPT III



C-C



A-A

FIGURE 4.3.2.11-1

ADAPTER STRUCTURAL ARRANGEMENT

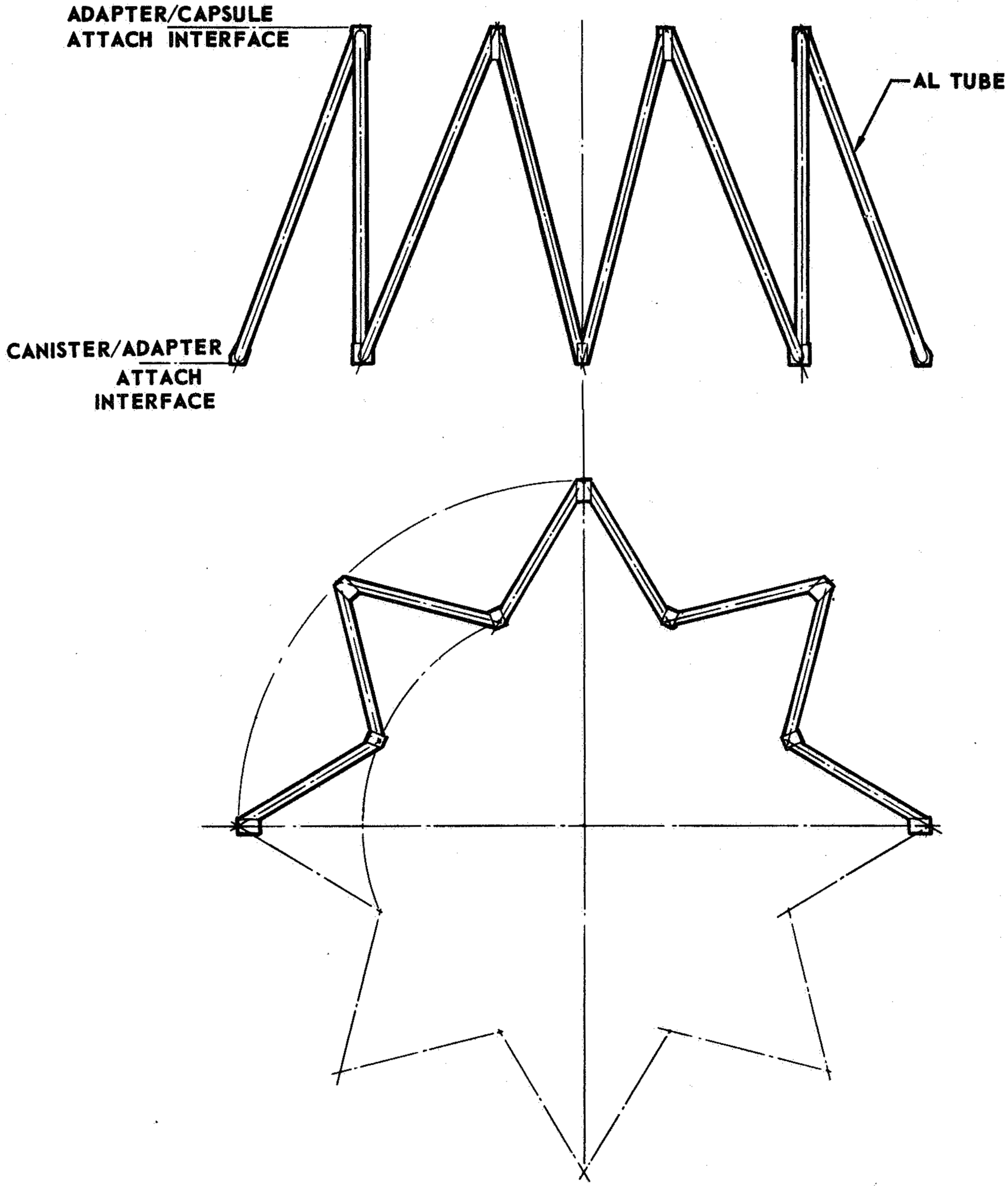


FIGURE 4.3.2.11–2

interface cone minimizes penetration, and therefore leakage, of the canister wall. Leakage is further minimized by sealing the structural joints during assembly with RTV-560, a room temperature-curing, elastomeric sealing compound.

Separation - Canister separation is accomplished at the field joing ring by detonating a dual confined explosive separation device (CESD - the device is redundant) which is sandwiched between the field joint ring flanges, as shown in Figure 4.3.2.11-3. The major elements of the separation system include the following:

a) CESD - This device consists of a mild detonating cord (MDC), positioned by a Teflon extrusion in a stainless steel tubing which is flattened to the shape of the groove in the field joint flange. The CESD is made up of two 180° sections, sealed at each end for each groove.

b) Stimulus Transfer Tube - This device contains an explosive for transferring the detonation to the CESD.

c) Electro-Explosive Device (EED)

d) Titanium Bolts - These bolts mechanically attach the forward and aft canister.

Upon receipt of an electrical signal the EED's are detonated, followed by detonation of the stimulus transfer tubes and then the CESD. Detonation of the CESD expands the steel tube and exerts a force between the clamped flanges of the field joint ring, which causes the bolts to fail in tension and the forward section of the canister to be ejected. The titanium bolts have their head end drilled to a depth which provides the desired breaking point and breaking strength. The broken bolts are constrained by a retainer from leaving the canister forward half, and the nuts are retained by nut plates. The canister separates at a velocity of from 1.25 to 10 fps.

Pressurization and Venting - The P&V equipment, as shown in the schematic of Figure 4.3.2.11-4, maintains the internal bio-integrity of the canister and controls internal pressure during terminal sterilization until capsule separation in Mars orbit. The major components and functions of the P&V equipment are:

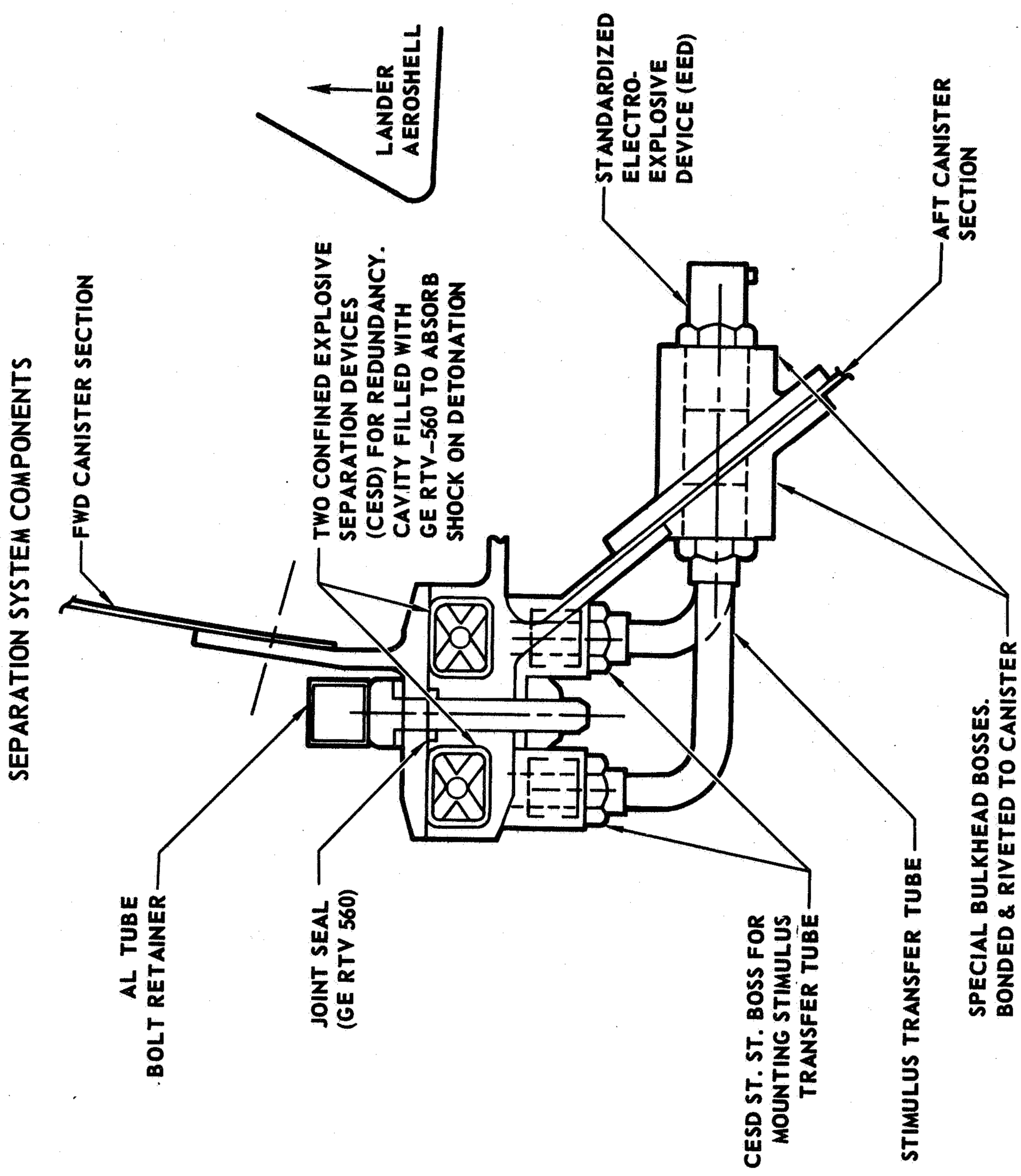


FIGURE 4.3.2.11-3

PRESSURIZATION & VENT SCHEMATIC

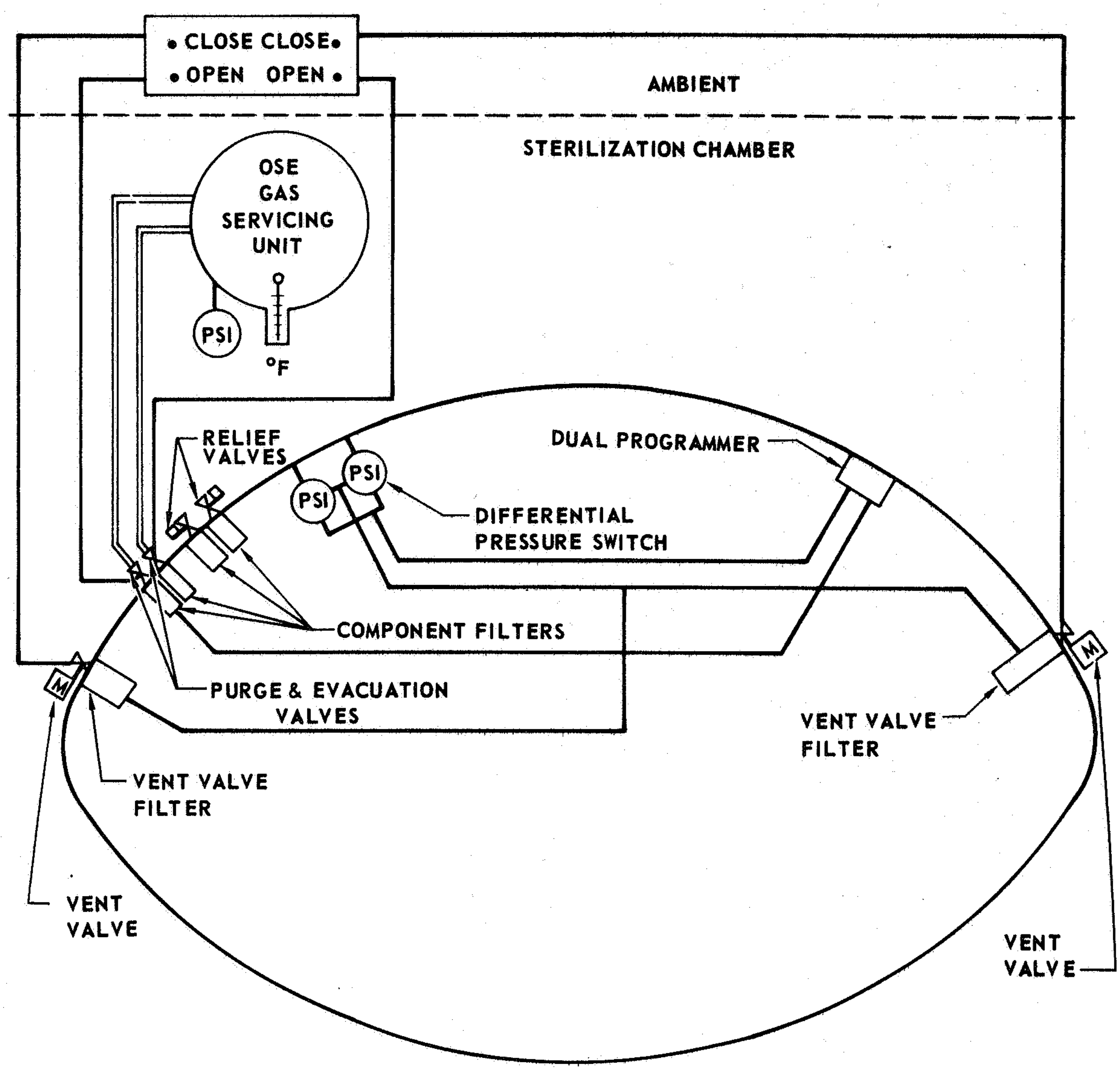


FIGURE 4.3.2.11–4

a) Vent Valves - Vent valves allow sterile gas to flow out of the canister during launch.

b) Relief Valves - Pressure operated relief valves protect the canister structure by limiting the internal pressure to 2.25 psi.

c) Purge and Evacuation Valves - These valves are used during decontamination and sterilization cycles on the ground. They are opened in Earth orbit and remain open to bleed off any residual pressure or outgassing.

d) Biological Filters - These filters are placed in series with each of the above valves to prevent recontamination of the canister interior if an inflow of gas through these valves occur.

e) Pressure Switches - Differential pressure switches provide a signal to the Operational Support Equipment (OSE) to reduce pressure on the ground and a signal after launch to close the vent valves when differential pressure falls below .50 psi.

Two each of the above major components are incorporated in the P&V system so that no single failure can have catastrophic results.

The OSE Gas Servicing Unit is connected to the purge and evacuation valves as soon as the canister is closed in preparation for terminal sterilization. It maintains internal pressure from .5 psi to 2.25 psi (limit pressure).

4.3.2.11.2 Adapter structure: The 17.3 lb adapter structure is a truss assembly consisting of 16 aluminum tubular members welded into an assembly which is attached to the capsule and canister at eight points, as shown in Figure 4.3.2.11-2. The truss-type design provides a short and stiff load path between the capsule and the canister. The adapter separates from the capsule when eight explosive bolts are detonated.

The adapter is designed to support the capsule for ground handling, launch, and cruise loads, and to allow for joining or separating the capsule and canister without access doors in the aeroshell or in the canister. This installation problem is accomplished by designing adapter-to-canister end fittings, with fixed plate nuts on the adapter side of the joint, to permit the adapter to be attached to the capsule. This assembly can then be installed within the canister aft section. Bolts which attach the adapter to the canister are installed from outside the canister with sealing washers, to complete the installation.

4.3.3 WEIGHT SUMMARY - The mass property data for Concept III is presented in Figures 4.3.3-1 thru 4.3.3-4. The Mission History Group Weight Summary, Figure 4.3.3-1, provides a weight breakdown by system at each essential point in the mission profile. This figure has the same format as the weight summary calculated by our automated weight program. The Center of Gravity (C.G.) and Inertia Summary, Figure 4.3.3-2, provides C.G. and inertia data for the modules and the total vehicle at essential points along the mission profile. This data is also presented in the same format as it is calculated by our automated weight program. The Surface Payload Weight Summary, Figure 4.3.3-3, is a breakdown by system of the landed payload weight. The Capsule Weight Summary, Figure 4.3.3-4, presents the data from Figure 4.3.3-1 in a different format.

CONCEPT III

MISSION HISTORY GROUP WEIGHT SUMMARY (POUNDS)

(BASED ON PARAMETRICS)

	CAPSULE WEIGHT	PRE-DEORBIT	POST-DEORBIT	ENTRY WEIGHT	LANDER WEIGHT	LANDED WEIGHT	SURFACE PAYLOAD
STERILIZATION CANISTER	190.0	0.0	0.0	0.0	0.0	0.0	0.0
ADAPTER (CAPSULE TO ORBITER)	17.3	0.0	0.0	0.0	0.0	0.0	0.0
DEORBIT PROPULSION	88.7	88.7	25.0	7.7	7.7	7.7	0.0
STRUCTURE – AEROSHELL	102.2	102.2	102.2	102.2	0.0	0.0	0.0
-INTERNAL	63.6	53.3	53.3	53.3	39.0	39.0	39.0
HEAT SHIELD	90.9	90.9	90.9	90.9	0.0	0.0	0.0
TEMPERATURE CONTROL SYSTEM	115.9	86.8	86.8	86.8	77.7	77.7	41.3
ATTITUDE CONTROL SYSTEM	41.3	40.1	38.2	37.7	0.0	0.0	0.0
GUIDANCE AND CONTROL SYSTEM	115.8	115.8	115.8	115.8	112.9	112.9	0.0
AERODYNAMIC DECELERATOR	45.5	45.5	45.5	45.5	0.0	0.0	0.0
TERMINAL PROPULSION SYSTEM	228.3	228.3	228.3	228.3	228.3	140.2	0.0
LANDING SYSTEM	138.3	138.3	138.3	138.3	138.3	138.3	0.0
GROSS PAYLOAD (1)	365.7	350.7	350.7	350.7	335.2	335.2	197.7
TOTAL	1603.5	1340.6	1274.9	1257.1	929.0	841.0	278.0

NOTE: (1) INCLUDES POWER AND SEQUENCER, TELECOMMUNICATIONS, SCIENCE EQUIPMENT, AND WIRING AND MOUNTING PROVISIONS.

FIGURE 4.3.3–1

CONCEPT III
C. G. AND INERTIA SUMMARY

INERTIA'S ABOUT INDIVIDUAL C.G.'S (SLUG-FT^2)			
LANDER (IMPACT) Z AXIS (ROLL)	69.92	X, Y AXIS (PITCH & YAW)	45.82
AEROSHELL Z AXIS	144.63	X, Y AXIS	77.65
LANDER + AERODECELERATOR Z AXIS	78.30	X, Y AXIS	58.54
ENTRY DEORBIT MOTOR X, Y, Z AXIS	0.09		
PRE-DEORBIT MOTOR X, Y, Z AXIS	0.71		
STERILIZATION CANISTER	179.50		
CENTER OF GRAVITY VALUES (FEET; + IS FWD. OF AEROSHELL BASE PLANE)			
STERILIZATION CANISTER	0.02	TOTAL CAPSULE	0.02
AEROSHELL	1.10	LANDER + AERODECEL	−0.03
DEORBIT MOTOR	−2.96	PRE-DEORBIT VEHICLE	0.02
ENTRY VEHICLE	0.21	LANDER (IMPACT)	0.08
INERTIA'S ABOUT VEHICLE C.G. (SLUG-FT^2)			
ENTRY VEHICLE Z AXIS	222.99		
PRE-DEORBIT VEHICLE Z AXIS	223.61		
TOTAL CAPSULE Z AXIS	403.11		
ENTRY VEHICLE X.Y AXIS	147.45		
PRE-DEORBIT VEHICLE X,Y AXIS	171.73		
TOTAL CAPSULE X, Y AXIS	351.23		

FIGURE 4.3.3–2

SURFACE PAYLOAD WEIGHT SUMMARY (POUNDS)

	PARAMETRIC (BASED ON SECTION 3.3.2)	POINT DESIGN BASED ON (SECTION 4.3.2)
POWER & SEQUENCER	(113.1)	(104.2)
EQUIPMENT	98.4	88.6
WIRING	14.7	15.6
TELECOMMUNICATIONS (INCLUDING INSTRUMENTATION)	(38.6)	(54.5)
EQUIPMENT	34.6	46.3
WIRING	4.0	8.2
SCIENCE	(34.4)	(35.5)
EQUIPMENT	30.0	30.0
WIRING	4.4	5.5
STRUCTURE	39.0	33.3
MOUNTING PROVISIONS	11.6	12.3
THERMAL CONTROL	41.3	49.6
TOTAL WEIGHT	278.0	289.4

FIGURE 4.3.3–3

CONCEPT III
CAPSULE WEIGHT SUMMARY (POUNDS)

	PARAMETRIC (BASED ON SECTION 3.3)	POINT DESIGN (BASED ON SECTION 4.3.2)
STERILIZATION CANISTER	190.0	190.0
ADAPTER	17.3	17.3
DEORBIT PROPULSION	((88.7))	((88.4))
PROPELLANT	63.7	68.1
INERT MOTOR	17.3	12.6
MISCELLANEOUS	7.7	7.7
AEROSHELL	((193.1))	((193.1))
STRUCTURE	102.2	102.2
HEAT SHIELD	90.9	90.9
STRUCTURE	((201.9))	((196.2))
INTERNAL	(24.6)	(24.6)
ADAPTER SEPARATION PROVISIONS	10.3	10.3
LANDER SEPARATION PROVISIONS	14.3	14.3
LANDING SYSTEM	138.3	138.3
SURFACE PAYLOAD	39.0	33.3
THERMAL CONTROL	((115.9))	((124.2))
CANISTER	29.1	29.1
AEROSHELL	4.5	4.5
THERMAL CURTAIN	14.6	14.6
LANDER	26.4	26.4
SURFACE PAYLOAD	41.3	49.6
FLIGHT CONTROL SYSTEM	((157.1))	((157.2))
ATTITUDE CONTROL SYSTEM	(41.3)	(37.6)
PROPELLANT	5.0	4.6
HARDWARE AND MOUNTING PROVISIONS	31.3	28.0
WIRING	5.0	5.0
ELECTRONICS	(115.8)	(119.6)
RADAR	55.1	57.1
GUIDANCE SYSTEM	37.0	37.0
WIRING AND MOUNTING PROVISIONS	23.7	25.5
PARACHUTE	((45.5))	((45.5))
CLOTH AND LINES	26.6	26.6
DEPLOYMENT, ATTACH, AND MISC	18.9	18.9
TERMINAL PROPULSION	((228.3))	((235.4))
PROPELLANT	99.3	98.0
GAS PRESSURANT	1.2	1.4
HARDWARE AND MOUNTING PROVISIONS	121.4	129.6 (1)
WIRING	6.4	6.4
POWER & SEQUENCER	((181.2))	((177.4))
EQUIPMENT	148.7	137.3
WIRING AND MOUNTING PROVISIONS	32.5	40.1
TELECOMMUNICATIONS (INCL. INSTRUMENTATION)	((117.2))	((100.0))
EQUIPMENT	91.5	74.7 (2)
WIRING AND MOUNTING PROVISIONS	25.7	25.3
SCIENCE	((67.3))	((69.2))
EQUIPMENT	53.0	53.0
WIRING AND MOUNTING PROVISIONS	14.3	16.2
TOTAL CAPSULE WEIGHT	1603.5	1593.9

NOTES: (1) USED EXISTING HARDWARE
(2) EQUIPMENT INTEGRATED – SPECIFIC DATA REQUIREMENTS RE-EVALUATED.

TABLE 4.3.3–4

4.4 CONCEPT IV

The 3500 lb Concept I capsule was planned to perform the maximum mission within the study constraints. The minimum mission consists of entry and surface atmospheric measurements, post-landed imaging and soil composition. The Concept IV payload also included descent television, a gas chromatograph, subsurface probe, and additional instruments for post-landed operations. Nominal environmental constraints and mission flexibility were assumed as the basis for this conceptual design.

The aeroshell diameter is 14.5 ft. AIDS and a three engine throttleable bipropellant propulsion system were selected to provide the final deceleration in order to deliver maximum payload within the diameter limit.

Work on Concept IV was discontinued to allow concentration on the other concepts.

"The aeronautical and space activities of the United States shall be conducted so as to contribute . . . to the expansion of human knowledge of phenomena in the atmosphere and space. The Administration shall provide for the widest practicable and appropriate dissemination of information concerning its activities and the results thereof."

—NATIONAL AERONAUTICS AND SPACE ACT OF 1958

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